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DATA REQUIREMENT MA-02
FINAL REPORT

TECHNICAL REPORT

ANALYSIS AND DESIGN

STUDY OF SOLID ROCKET MOTORS FOR A SPACE SHUTTLE BOOSTER

CONTRACT NO. NAS8–28429
JANUARY 13, 1972 TO MARCH 15, 1972

MARCH 15, 1972

PREPARED FOR
THE NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
MARSHALL SPACE FLIGHT CENTER, ALABAMA 35812

LOCKHEED PROPULSION COMPANY
P.O. BOX 111 REDLANDS, CALIFORNIA 92373

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A. H. Von Der Esch
Lockheed Propulsion Company
Vice President, Technical and Marketing

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ABSTRACT

Lockheed Propulsion Company conducted an analyses and design effort as part of the Study of Solid Rocket Motor For A Space Shuttle Booster.

Lockheed Propulsion Company selected the 156-inch-diameter, parallel-burn Solid Rocket Motor as its baseline because it is transportable and is the most cost-effective, reliable system that has been developed and demonstrated. The basic approach taken by LPC in this study was to concentrate on the selected baseline design, and to draw from the baseline sufficient data to describe the alternate approaches also studied.

As a result of the study, Lockheed Propulsion Company reached the following conclusions with respect to technical feasibility of the use of solid rocket booster motors for the Space Shuttle Vehicle:

(1) LPC's 156-inch, parallel-burn baseline SRM design meets NASA's study requirements while incorporating conservative safety factors.

(2) The Solid Rocket Motor Booster represents a cost-effective approach.

(3) Baseline costs are conservative and are based on a demonstrated design.

(4) Recovery and reuse are feasible and offer substantial cost savings.

(5) Abort can be accomplished successfully.

(6) Ecological effects are acceptable.

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FOREWORD

This document is Book 1, Analysis and Design of Volume II, Technical Report. It is a part of Lockheed Propulsion Company's final report for the Study of Solid Rocket Motors for a Space Shuttle Booster. The final report consists of the following documents:

Volume I	Executive Summary
Volume II	Technical Report
Book 1	Analysis and Design
Book 2	Supporting Research and Technology
Book 3	Cost Estimating Data
Volume III	Program Acquisition Planning
Volume IV	Mass Properties Report

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SUMMARY

Lockheed Propulsion Company's objective from the outset of the Space Shuttle Program has been to provide complete and conservative design and cost parameters for an expendable Solid Rocket Motor (SRM) Booster Vehicle for the Space Shuttle Program. With this approach, LPC has attempted to identify the maximum technical and cost risks that could be encountered by NASA in employing a solid rocket motor as the Space Shuttle Booster Vehicle. Therefore, LPC believes that the baseline vehicle costs presented in this report are distinctly conservative and will be reduced upon further definition and detailed estimating. Two items, which LPC has not included and which will affect a fixed-payload program cost, are escalation and profit, both of which were directed in the Study Contract to be deleted from consideration.

As directed by NASA, LPC also attempted to determine "hard" versus "soft" costs, and an upper band was established above the baseline for a "worst condition." As a result of Lockheed's solid rocket motor experience, the propulsion system costs are "hard" and, therefore, an upper limit of 2 percent on the SRM cost has been defined. LPC believes that the Stage costs are "soft" and a 30-percent upper limit on the Stage cost was established. With the SRM and Stage combined, a total of 10-percent upward variation has been identified in the Booster Vehicle (WBS 3.3) Program costs. A lower range has also been established, which identifies potential reductions for thrust vector control, thrust termination, and recovery.

The Booster Vehicle selected as the baseline configuration is a parallel-burn (two-motor) 156-inch-diameter SRM vehicle sized for the large (65,000-pound) Orbiter payload. The baseline program assumed for study purposes includes a 5-year (1973 - 1978) development/qualification program, a 13-year (1976 - 1988) production program, and an 11-year (1978 - 1988), 440 vehicle launch program.

The development program includes 25 SRMs; 5 development motor tests, 4 PFRT motor tests, 2 inert booster vehicles (2 SRMs per vehicle) and 6 launches (1 unmanned and 5 manned flights with 2 SRMs per vehicle). All 25 motors in the development program will be fabricated in LPC's existing, large-motor Potrero manufacturing facility. The development program schedule was established at 5 years to minimize annual funding and could be shortened by as much as 1 year without impacting the launch schedule.

The production program of 440 launches includes manufacture of 883 SRMs (880 for launches and 3 for production facility start-up demonstration) and 440 sets of Stage hardware. Due to the nature of the solid rocket motor, quality is ensured by the facility process controls in manufacturing. Thus a three-motor test program is planned to demonstrate that the production facilities will reproducibly deliver the SRMs qualified during development. As directed in the Study Contract, all launches were considered to be from Kennedy Space Center (KSC).

Lockheed Propulsion Company, as prime contractor for the Booster Vehicle, would utilize all of the industry production capability before additional facility expansion. LPC would subcontract to at least two other SRM manufacturers for a portion of the production motors. Additionally, all components would be considered for dual procurement to ensure a redundant capability for Booster Vehicle delivery. This LPC plan provides Booster Vehicle procurement at a very low risk to NASA in the event of a labor, facility, or material problem at any time during the program. This approach also results in a relatively low facility expansion cost ($25.7 million) for the production program and avoids the building of a brand new facility, which would cost approximately $70 million.

The three production facility start-up demonstration tests are considered adequate by LPC to qualify the three production facilities (LPC and two others) for the baseline costing effort. It was considered that NASA might desire additional testing to qualify the new subcontractors ("second sources") and, therefore, nine motor tests were included in establishing the upper limit 2-percent variation in SRM costing. However, LPC recommends only three tests and has used this in the baseline costing.

Previously, it has been stated that the baseline design is conservative. As evidence of this, all metal structures have a minimum safety factor of 1.4. This has naturally imposed an additional cost on materials, but LPC believes that this should be maintained, thus guaranteeing the high reliability required for a man-rated system. As a bonus feature, analysis indicates that the motor chamber with this safety factor (wall thickness 0.460 inch) will withstand water impact loads at 100 feet per second and at entrance angles up to 45 degrees. Although recovery/reuse is not considered in the baseline costing, Lockheed's SRM design should therefore not require additional strengthening (higher material costs) should recovery/reuse prove cost-effective for the Booster Vehicle.

As further evidence of a conservative design, the safety factor for all ablative insulation materials was established at 2.0. Once again, it is felt that this should be maintained for man-rated reliability. In the areas of thrust termination (TT) and thrust vector control (TVC), no firm requirement was established by either the Phase B contractors or by the customer. LPC assumed that the Booster Vehicle would require both TT and TVC, plus a strenuous TVC duty cycle, which sized the system conservatively.

The baseline costs are backed by firm vendor quotes on procured components and conservative labor estimates. Lockheed's labor estimates were prepared from a task definition or "ground-up" standpoint, based on previous LPC large-motor experience, other LPC rocket motor programs, and also on related industry experience on solid propellant rocket motors. Nine full-scale, 156-inch-diameter demonstration motors have been test-fired to date, five by Lockheed Propulsion Company. These tests are summarized in the following table.

SUMMARY OF 156-INCH LARGE SOLID ROCKET MOTOR TESTS

No.	Date	Motor Description: Designation	Motor Description: Fabrication	Test Data: Maximum Thrust (lb)	Test Data: Average Thrust (lb)
1.	1964 May	156-3	LPC	0.95M	0.88M
2.	Sep	156-4	LPC	1.09M	1.00M
3.	1965 Feb	156-2C-1	TCC	3.25M	2.97M
4.	Dec	156-1	TCC	1.47M	1.29M
5.	Dec	156-5	LPC	3.11M	2.84M
6.	1966 Jan	156-6	LPC	1.03M	0.94M
7.	Apr	L-73	LPC	0.66M	0.60M
8.	May	156-7	TCC	0.39M	0.32M
9.	May	156-9	TCC	0.98M	0.88M

All of these motors, with thrust levels up to three million pounds, performed within 2 percent of their calculated parameters, and only one incident (involving the loss of an exit cone in a moveable nozzle test by another contractor) was experienced. This is a significant feat in that each of the nine motors was a "one-of-a-kind" configuration and involved reuse of LPC-designed case hardware as many as four times. Lockheed is proud of this 100-percent successful completion of its five 156-inch motor tests, which were accomplished under-budget on firm fixed price contracts (see USAF Testimonials in Appendix A of the Cost Book).

As previously stated, the experience gained in these programs was applied by all LPC branches in estimating the labor for the Booster Vehicle. In the area of motor processing, the hands-on-hardware "first-unit" labor hours for the baseline were estimated, and then a 90-percent labor improvement or learning curve was applied. Comparison with both LPC experience and other SRM industry experience indicates that this is conservative; in the majority of previous programs, improvement curves in the middle to low eighties have been experienced. For example, on the basis of two large weapon systems, Minuteman and Poseidon, an improvement curve in the 80- to 85-percent range should be achievable in the Booster Vehicle. For this additional reason, LPC, employing a 90-percent curve, has estimated the baseline configuration production costs in a conservative manner.

As another consideration in development of the costs, LPC began this study on 13 January 1972 assuming that the Booster System (WBS 3.0) was to be costed. On 2 February, LPC was notified that the SRM contractors were to price at the Booster Vehicle level (WBS 3.3). While this was intended by NASA to alleviate the SRM contractors' efforts in the short study time available, it did turn out to add another variable, which is reflected as additional conservatism in the LPC costs. Included in LPC's costs are some items that could be interpreted as belonging under Booster Management (WBS 3.1), System Engineering (WBS 3.2), or Booster System Support (WBS 3.5), which may not be included in the cost estimates of the other study contractors.

The Booster Vehicle program costs (WBS 3.3) presented by LPC on 14 and 23 February 1972 were based on the previously defined configuration and costing assumptions. The LPC baseline Booster Vehicle cost estimate presented on these dates is summarized below.

	SRM	Stage	Total Booster Vehicle
Development	$ 141.6M	$ 48.2M	$ 189.8M
Production	2,545.7M	929.0M	3,474.7M
	$2,687.3M	$977.2M	$3,664.5M
Total Program Cost/Launch	$ 6.0M	$ 2.2M	$ 8.2M
Recurring Cost/Launch	$ 5.8M	$ 2.0M	$ 7.8M

The total program cost per launch is developed by dividing the total program cost (3,664.5 million) by the total number of manned launches (445). Although cost per launch does not normally include amortization of DDT&E or non-recurring production items, LPC chose to attempt to display the total program liability that NASA could encounter in employing a solid rocket motor Booster Vehicle. The standard way of displaying cost per launch is by using the recurring unit cost, which, for LPC's baseline, is $7.8M. Once again, these program costs were developed early in the Study Program with the objective of identifying the maximum technical and cost risk that could be encountered by NASA.

On 12 February, after the cut-off date for the 14 and 23 February presentations, Lockheed began a second iteration of the program baseline configuration and cost. Labor and material were analyzed in more depth, more definition was prepared to separate recurring from nonrecurring costs, and the Operations portions of the SRM and Stage were separated into more identifiable activities. This resulted in a redistribution of the baseline costs as shown in the following two tables:

	SRM	Stage	Operations	Total
Development	$ 131.0M	$ 31.0M	$ 27.8M	$ 189.8M
Production	~~2,434.9M~~ 2,303.9M	626.5M	544.3M	3,474.7M
	$2,434.9M	$657.5M	$572.1M	$3,664.5M

Note that in both tables the previously shown total program costs have remained unchanged but are redistributed by LPC for better understanding.

	Total Costs	Recurring Cost/Launch	Total Cost/Launch
Recurring SRM production	$2,242.8M	$5.1M	$5.1M
Recurring Stage production	626.5M	1.4M	1.4M
Recurring operations	544.3M	1.2M	1.2M
Nonrecurring production	61.1M	0	0.1M
Development	189.8M	0	0.4M
Total	$3,664.5M	$7.7M[(a)]	$8.2M

The next step in the second iteration of the baseline configuration and cost was to review areas where cost might be overly conservative and could thus be reduced. Since the hardware is a major portion of the SRM cost, additional definition and breakdown of vendor component and material costs were requested from the subcontract suppliers. In vehicle configuration, better design definition was developed and rebids were prepared in some areas. As an example, in January, prior to completion of the TVC system sizing, quotes had to be obtained on the actuator. LPC requested bids on the actuator used on the S1-C Vehicle, knowing that it would be more than adequate for the job. The actuator requirement was found to be far less and was rebid at a significantly lower cost. Safety factors of all hardware were maintained and the material costs still reflect safety factors of 1.4 on structures and 2.0 on ablative insulations.

The motor processing tasks and the improvement/learning curve were reviewed in considerable depth. A steeper curve (86 percent) was selected as realistic but still sufficiently conservative in comparison to other major solid rocket motor programs and LPC's 156-inch motor experience. Assembly and support labor were also analyzed and some areas of redundancy between WBS paragraphs were identified and deleted. The analysis of labor and material on the SRM has resulted in a lower unit cost position for the SRM baseline. These analyses have been time-consuming and, although some areas of the Stage attachment hardware and Operations have been reviewed and reduced, additional effort is being expended by Lockheed toward further definition, analysis, and reduction.

To support a final report date of 15 March, a cut-off was made on 8 March in the second costing iteration. The reduced program costs are shown in the following table as "Baseline, Revision 1" and are compared by item to the original baseline costs shown previously.

(a) As a minor note, the redistribution identified additional nonrecurring production costs, resulting in a lower recurring cost per launch.

	Baseline Cost	Reduction	Baseline Revision 1
Recurring SRM Production	$2,242.8M	$266.8M	$1,976.0M
Recurring Stage Production	626.5M	155.7M	470.8M
Recurring Operations	544.3M	98.0M	446.3M
Nonrecurring Production	61.1M	0	61.1M
Development	189.8M	3.7M	186.1M
	$3,664.5M	$524.2M	$3,140.3M
Total Cost/Launch	$ 8.2M	$ 1.1M	$ 7.1M
Recurring Cost/Launch	$ 7.7M	$ 1.1M	$ 6.6M

Each of the reductions shown in this table is discussed in the Addendum to the cost book of the final report. The cost per launch, both recurring and total, has been reduced by over a million dollars. Further analysis will yield even more reductions in the areas of Stage and Operations. It is believed by Lockheed that the SRM, however, will not yield further major reductions without a change in either performance or hardware safety factors, which is not recommended by LPC.

Therefore, the Baseline Revision 1 costs ($3,140.3B) are submitted as Lockheed's formal position on the SRM Booster Vehicle (WBS 3.3).

The conclusions of the LPC study are:

(1) The LPC 156-inch-diameter baseline design meets all the technical requirements for the Booster Vehicle.

(2) The baseline design appears to have the structural capability to withstand recovery-load impacts should recovery/reuse prove cost-effective for the Booster Vehicle.

(3) The SRM Booster Vehicle, because of its demonstrated technology, can be developed to meet all NASA schedule requirements.

(4) The Baseline Revision 1 costs are realistic and achievable and are subject to further reduction.

(5) The cost for development ($186.1M) of an expendable SRM Booster Vehicle are less than 4.0 percent of the total Space Shuttle Development budget ($5.5B).

(6) The Baseline Revision 1 SRM Booster Vehicle cost per launch (recurring $6.6M, total $7.1M) is less expensive than that of a liquid booster.

In summary, Lockheed believes that an SRM propulsion system can perform the mission, can be easily developed in the time available, and will prove to be a cost-effective booster vehicle for the Space Shuttle Program.

Section 1

INTRODUCTION AND SUMMARY

Lockheed Propulsion Company (LPC) has conducted an analysis and design effort as part of the Study of Solid Rocket Motors for a Space Shuttle Booster. This effort was directed to the following technical requirements established by National Aeronautics and Space Administration (NASA):

- Orbiter payload: 45,000 and 65,000 pounds
- Parallel- and series-burn 120- and 156-inch diameter Solid Rocket Motors (SRMs), with and without thrust vector control (TVC) and thrust termination (TT)
- Stage requirements to be obtained from Phase B system contractors.

In addition to these NASA-specified study requirements, LPC imposed additional groundrules on itself for the conduct of the study; i.e., the baseline approach must be representative of the results of the study effort conducted by the Phase B prime contractors, and all selected design and fabrication features of the baseline SRM must incorporate proven technology, with strong emphasis on high reliability.

The primary goal of LPC's technical effort has been to identify the most likely Solid Rocket Motor for the Space Shuttle Booster. The selection was based on the following:

- LPC's substantial large solid rocket motor experience
- Vendor experience
- Related or prime contractor experience
- Other solid propellant industry experience

The basic approach taken by LPC in this study has been to concentrate on a single baseline design, and essentially to draw from this baseline sufficient data to describe the study alternates.

The 156-inch, parallel-burn SRM was selected as LPC's baseline because (1) it is a developed and demonstrated design for which LPC has accumulated a background of credible cost information, (2) it is a readily transportable system, (3) it is the most cost-effective approach, and (4) it responds to the NASA request that the parallel-burn configuration be given primary emphasis. The illustration below summarizes LPC's conservative approach to the study.

SUMMARY

WHAT — SRM BASELINE COSTS ARE CONSERVATIVE

WHY — CONSERVATIVE IN DESIGN APPROACH
- 1.4 SAFETY FACTOR - PRESSURE VESSEL
- 2.0 SAFETY FACTOR - INSULATION

USE OF ACTUAL COST HISTORY

UPDATED SUPPLIER INFORMATION

COSTS BASED ON DEMONSTRATED TECHNOLOGY

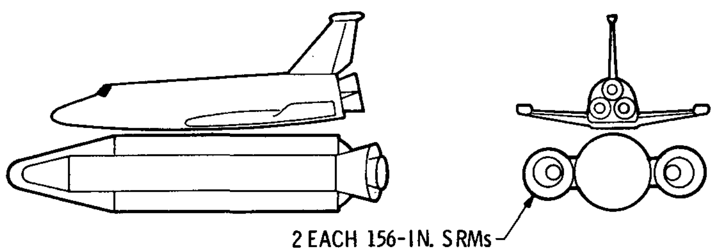
COSTS INCLUDE
- THRUST VECTOR CONTROL
- THRUST TERMINATION
- EXPENDABLE BOOSTER

It can be seen that LPC chose to be conservative with respect to both design and costing approaches, although the specified study groundrules permitted the contractors wide latitude.

A total of nine 156-inch motors in various configurations have been statically test-fired in the past 10 years. All have been successful. Lockheed Propulsion Company test-fired five of these motors.

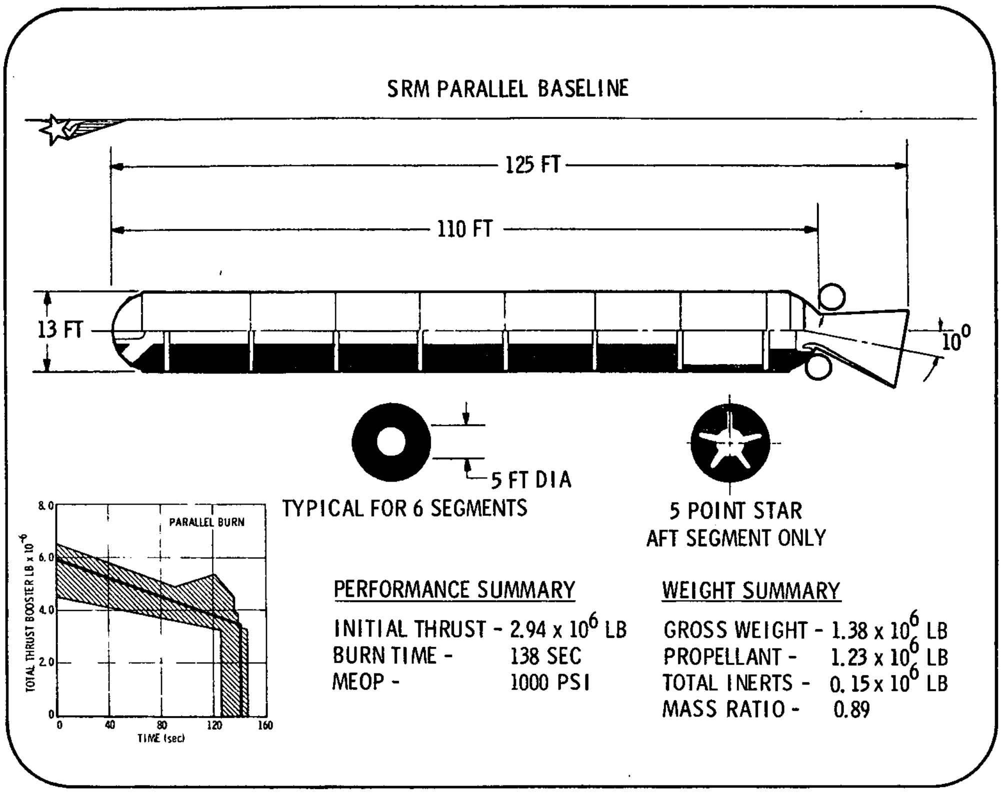
The baseline parallel-burn vehicle configuration is shown below. It is representative of the configurations and vehicle weights received from the Phase B prime contractors. The booster lift-off weight of 2.835 million pounds is compatible with the 65,000-pound payload.

156-INCH VEHICLE BASELINE
PARALLEL BURN



WEIGHTS LB x 10^{-6}	
ORBITER LIFT-OFF WEIGHT	1.800
BOOSTER LIFT-OFF WEIGHT	2.835
GROSS LIFT-OFF WEIGHT	4.635

The illustration below shows the general configuration, performance, and weight information for the selected baseline motor, a 156-inch, parallel-burn, 7-center-segment SRM.



Each motor generates an initial thrust of 2.94 million pounds, has a burn time of 138 seconds, and operates at a maximum expected operating pressure of 1000 pounds per square inch. Propellant weight is 1.23 million pounds, total motor weight is 1.38 million pounds, and the motor mass fraction is 0.89. The inert weights and mass fraction presented here are considered to be conservative. They include the weights of thrust termination and thrust vector control systems designed to meet a most severe set of requirements. It is estimated that the motor can be optimized to achieve a mass fraction of greater than 0.90 with the thrust vector control and thrust termination systems included, or greater than 0.91 if these systems are excluded.

The baseline thrust-time curve is shown in the middle of the cross-hatched area in the previous figure. The cross-hatched band represents the range of Phase B prime contractor inputs. Motor performance can be tailored to match any of the specific prime contractor requirements.

The selected components for the baseline SRM are as follows:

Component	Approach	Reason for Selection
Motor case	D6AC steel, 225 Ksi ultimate strength	Extensive production experience - Minuteman
Nozzle	Ablative plastic throat	Low risk, materials proven
Igniter	Head-end pyrogen	Conventional SRM approach
Internal insulation	Filled NBR sheet stock, autoclave-cured	Proven reliability
Propellant	PBAN, LPC-580, Modified, Class II	Demonstrated on 156-inch SRMs
Thrust termination system	Dual, head-end ports	Used on Poseidon, Minuteman, and Titan III
Thrust vector control system	Lockseal flexible joint	100 successful flights -- used on Poseidon

The basis of selection for the baseline SRM components is demonstrated experience. This approach provides for minimum-risk booster development and the availability of cost information based on actual experience. Each of the components has an extensive production history. The propellant, a polybutadiene acrylonitrile (PBAN), was used in previous 156-inch motors fired at Lockheed Propulsion Company. This propellant is safe to handle and has been classified as nondetonable.

A more detailed summary of the features of the baseline SRM is shown on the following page.

The key stage features are shown below. Conventional attachment and separation methods are incorporated in the design. The electrical characteristics are also straightforward, with emphasis on safety and high reliability.

FEATURES OF BASELINE SRM

Parallel Burn Booster

Seven-segment, 156-inch SRM with Thrust Vector Control and Thrust Termination

DESIGN CHARACTERISTICS AND DIMENSIONS

PROPELLANT AND GRAINS

- Propellant type: PBAN (LPC-580 Modified)
- Propellant total weight: 1.23×10^6 pounds
- Motor MEOP: 1000 psi
- Propellant characteristics
  - I_{sp} std: 262.6 seconds
  - Density: 0.0646 lb/in.^{3}
  - Burn rate at 1000 psi: 0.395 in./sec
  - Burn rate exponent: 0.3
- Grain configurations
  - Circular port in forward and aft head sections and first six cylindrical segments. 5-point star in aft cylindrical segment.
- Grain segment length: 158 inches
- Nominal grain port diameter: 59.65 inches
- Nominal grain OD: 154.68 inches
- Nominal web thickness: 47.51 inches

IGNITER

- Type: Pyrogen
- Case material: D6AC
- Propellant: LPC-580A
  - Grain weight: 500 pounds
  - Burn time: 0.5 second
- Insulation: asbestos-filled NBR
- Other
  - EBW type dual initiators

CASE SEGMENTS

- Material: D6AC steel
- Strength level: UTS 225
- Biaxial gain: 13 percent
- Fabrication method
  - Roll formed/machined joints/no welds
  - L/D = 1:1.026
  - 7 segments plus domes
- Joint configuration
  - Pin type: Tapered
  - Seal type: Barrel O-ring
- Nozzle attachment
  - Canted nozzle flange preferred
  - Alternate bolt-on adapter
- Case wall thickness: 0.460 inch

DOME SECTIONS (Common Fore and Aft)

- Material: D6AC
- Strength level: UTS 225
- Biaxial gain: 13 percent
- Fabrication method
  - Roll formed, swaged/machined joints/no welds
  - Integral skirt, if possible with forging restrictions
  - Alternate: Bolt-on skirt
  - Two TT ports canted 45 degrees

INTERNAL INSULATION

- Material
  - Aft closure: Silica- and asbestos-filled NBR, e. g., Gen-Gard V-44 or equivalent
  - Segments and forward dome: Silica-filled NBR, e. g., Gen-Gard V-45 or equivalent
- Installation
  - Sheet stock layup
  - Cure in place
- Alternate being costed is mastic/cast insulation

NOZZLE

- Submerged entrance (standard practice for movable nozzles)
- All ablative parts tape-wrapped
  - Carbon-phenolic
  - Silica-phenolic
  - Glass-phenolic
- Glass overwrap exit cone structure
- All steel parts - D6AC
- Inhibitors
  - As required
  - Silica-filled NBR
- Throat area: 2150 in.^{2}
- Throat diameter: 52.32 in.
- Expansion ratio: 8.33:1
- Half angle: 17.5 deg

TVC

- N_2 cold-gas blowdown type system
- Two hydraulic servo-actuators (90 degrees apart)
- Actuators are linear double-acting

TOTAL INERT WEIGHT

- 154×10^3 pounds

PERFORMANCE

SRM BALLISTIC PARAMETERS (80^{0}F, sea level)

- Burn time: 138 seconds
- Average chamber pressure: 631 psi
- Maximum expected chamber pressure: 1000 psi
- Predicted vacuum thrust coefficient: 1.6671
- Initial vacuum thrust: 2,942,351 lbf
- Initial sea level thrust: 2,679,163 lbf
- Specific vacuum impulse, $\epsilon = 8.33$, $\alpha = 17.5^0$: 264.8 lbf-sec/lbm
- Total vacuum impulse: 326.01×10^6 lbf-sec

TVC OPERATING CHARACTERISTICS

- 10-degree deflection
- 15 deg/sec slew rate
- Design stall torque 16×10^6 in.-lb
- System has capability for 20 full deflection cycles

THRUST TERMINATION OPERATING CHARACTERISTICS

- Function time: 2 milliseconds
- Function time tolerance between ports: 4 milliseconds
- Time to equilibrium pressure: 1.0 second

WEIGHT SUMMARY

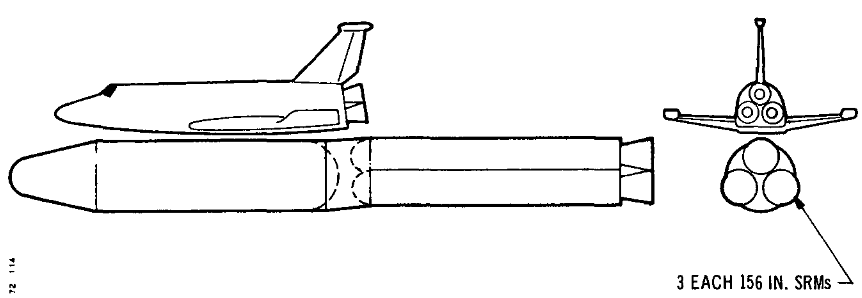
Booster (at lift off): 2,834,586 pounds
Interstage: 65,000 pounds
SRM: 1,384,793 pounds

- Motor case: 95,373 pounds
- Insulation and liner: 13,971 pounds
- Nozzle: 17,004 pounds
- Igniter: 1000 pounds
- Thrust termination: 7,915 pounds
- Thrust vector control system: 18,500 pounds
- Propellant: 1,231,030 pounds
- Motor mass ratio: 0.889

BASELINE STAGE FEATURES

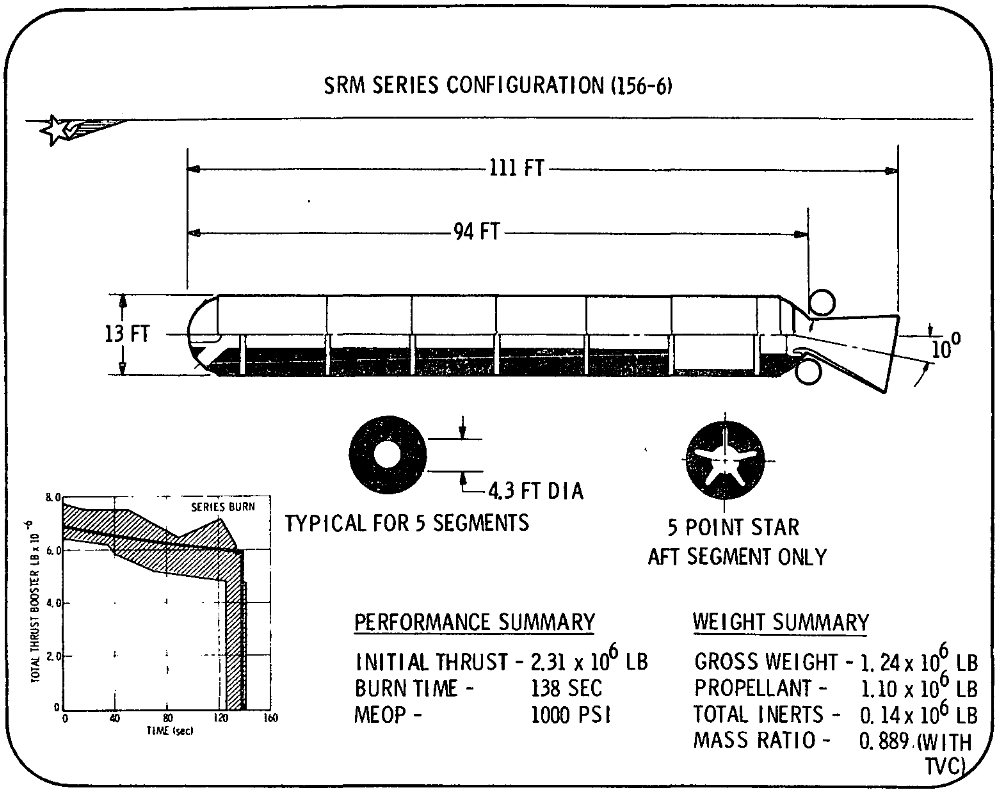
System	Approach	Reason for Selection
Mechanical	Thrust take-out forward on centerline	Distribution of loads
	Flared aft skirt	Reduced nozzle torque
	Small solid motors for SRM separation	Reliable, positive separation force
Electrical	No raceway; umbilical to orbiter	Simplicity, cost
	EBW high-voltage initiation	Safety, reliability
	Redundant circuitry and power	Reliability

Alternate solid rocket motor designs. The 6-segment series-burn SRM differs only slightly from the parallel-burn configuration. Two 7-segment, 156-inch SRM units are used in the parallel configuration, whereas three 6-segment, 156-inch SRM units comprise the series-burn design. The illustration below shows the configuration of the 156-inch series-burn vehicle.



	WEIGHT (LB X 10^{-6})
ORBITER LIFT-OFF WEIGHT	1.25
BOOSTER LIFT-OFF WEIGHT	3.82
GROSS LIFT-OFF WEIGHT	5.07

General motor configuration, performance, and a weight summary are shown below.



A detailed design was not made specifically for the 120-inch SRM alternate. Details of the United Technology Center No. 1207 Titan booster motor were used for this purpose. The primary reason that LPC chose not to make a special 120-inch design for the SRM is that this motor has been designed and developed. Emphasis was placed on the 156-inch size because fewer units are required to perform the mission, resulting in significantly lower costs.

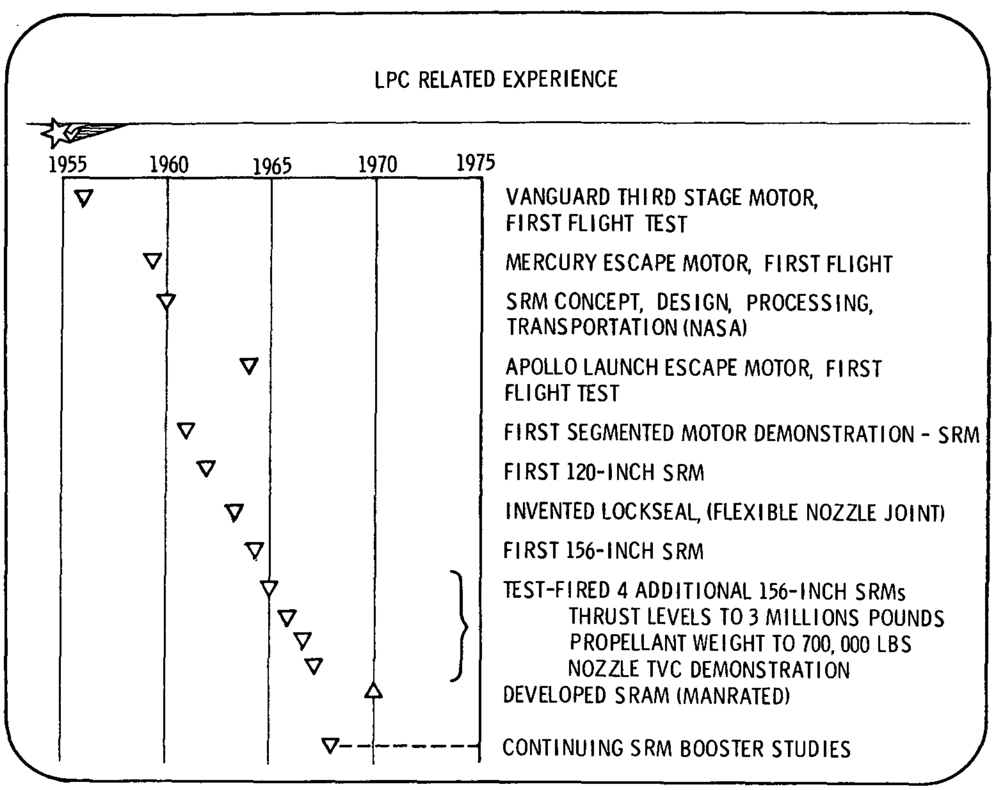
Key issues. Lockheed Propulsion Company also evaluated the key issues related to the SRM booster: recovery/reuse, abort, and ecological considerations.

Recovery/reuse. Although all of LPC's cost data are based on a fully expendable solid rocket motor, a recovery system for solid rocket motors appears to be feasible. If such a system were developed, a significant savings in cost per launch can be achieved over the expendable baseline program cost.

Abort. The solid rocket motor recommended for the Space Shuttle Booster can provide assurance of safe abort. An independent orbiter escape system will be required for potential use during the early critical phase of flight (considered to be during the first 40 seconds of flight).

Ecology. The current impact of ecology considerations is well recognized. From a technical standpoint, LPC's research and analysis indicates that serious problems do not exist. Waste disposal and noise present no ecology problems. The only possibility of a potential problem is the generation of very dilute hydrochloric acid in the atmosphere if the launch were to occur during conditions of extremely high humidity (such as during a heavy rain storm) and on-shore winds. Even if these conditions were encountered, there would be no adverse effect on personnel and only minor effects (even in the immediate area) on plant life.

Lockheed Propulsion Company has played a very significant role in the development of large solid motor technology. Among other achievements, LPC designed, built, and test-fired the first 120-inch-diameter solid rocket motor in 1962 and the first 156-inch-diameter motors in 1964. Lockheed Propulsion Company has manufactured and fired five of the nine 156-inch SRMs tested to date. LPC also has a strong background in man-rated rocket motors, such as the Escape Motor for the Mercury Capsule, the Apollo Launch Escape Motor, and the sophisticated two-pulse motor for the air-launched SRAM missile. Programs such as these have given LPC a depth of know-how in the conduct of programs managed under stringent controls. The following figure summarizes pertinent LPC experience.



In summary, Lockheed Propulsion Company's study, based on a strong background of SRM and other pertinent experience, has resulted in the following conclusions with respect to feasibility of the use of solid rocket booster motors for the Space Shuttle Vehicle:

TECHNICAL

- The baseline design meets NASA requirements.
- Recovery and reuse are feasible.
- Abort can be accomplished successfully.
- Ecological effects are acceptable.

SCHEDULE

- Schedule milestones are realistic.

COST

- Baseline costs are conservative, based on demonstrated experience.
- The SRM booster is cost-effective.
- Recovery and reuse has the most significant potential for further cost reduction.

Section 2

CONFIGURATION PRELIMINARY DESIGN

In this study, LPC elected to concentrate on a single baseline design that was representative of the results of the study efforts received from the Phase B prime contractors. This baseline design also served to generate the data required to describe the alternate designs requested. The selected parallel-burn baseline design, which incorporates the most credible information available, uses two seven-segment 156-inch-diameter units. Design details and performance characteristics for this baseline are shown in subsection 2.1.

The alternate, series-burn 156-inch, six-segment SRM design (presented in subsection 2.2.1) differs only slightly from the baseline, but requires three 156-6 SRM units.

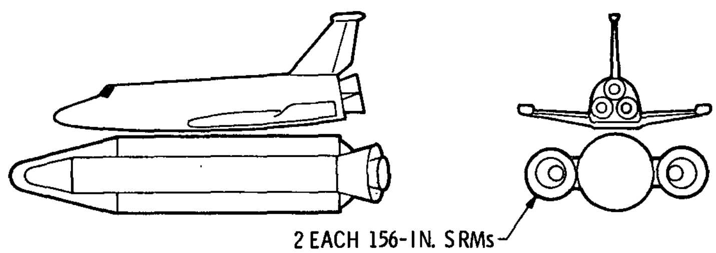
Motors with diameters larger than 156 inches were not evaluated in depth for the parallel-burn SRM baseline because larger sizes do not offer the combination of advantages provided by the selected baseline.

Lockheed Propulsion Company chose not to prepare a new design for the 120-inch SRM because of the significant advantages shown by the 156-inch motor in prior industry studies, and because the 120-inch motor is fully developed and demonstrated. Instead, details of the United Technology Center No. 1207 Titan Booster Motor were used for design purposes. This design is shown in subsection 2.2.2.

2.1 BASELINE DESIGN: 156-INCH PARALLEL-BURN SRM

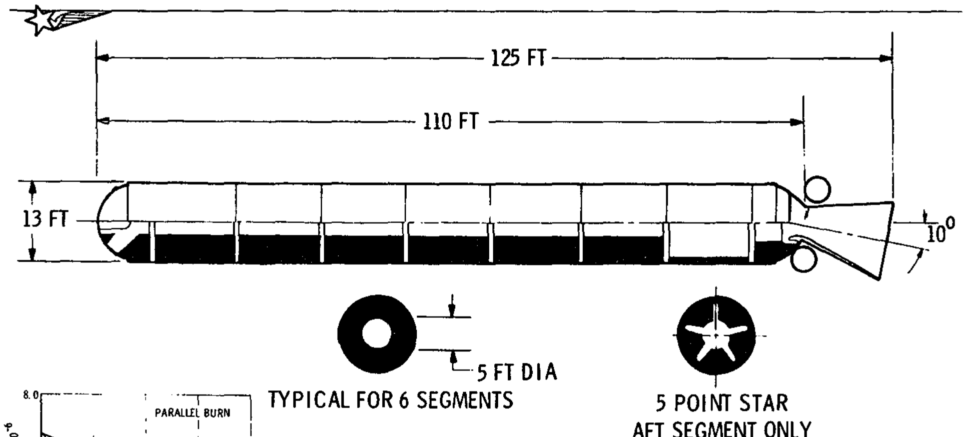
2.1.1 Design

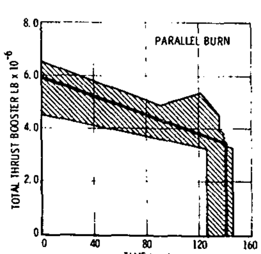
156-INCH VEHICLE BASELINE
PARALLEL BURN



WEIGHTS LB x 10^{-6}	
ORBITER LIFT-OFF WEIGHT	1.800
BOOSTER LIFT-OFF WEIGHT	2.835
GROSS LIFT-OFF WEIGHT	4.635

SRM PARALLEL BASELINE





PERFORMANCE SUMMARY

INITIAL THRUST - 2.94 x 10^6 LB
BURN TIME - 138 SEC
MEOP - 1000 PSI

WEIGHT SUMMARY

GROSS WEIGHT - 1.38 x 10^6 LB
PROPELLANT - 1.23 x 10^6 LB
TOTAL INERTS - 0.15 x 10^6 LB
MASS RATIO - 0.89

SRM BASELINE CONFIGURATION

CASE SEGMENTS

- Material: D6AC
- Strength level: UTS 225 Ksi
- Biaxial gain: 13 percent
- Fabrication method
  - Roll formed/machined joints/no welds
  - L/D = 1:1
  - Parallel: 7 Segments plus domes
- Joint configuration
  - Pin type: Tapered
  - Seal type: Barrel O-ring
- Nozzle attachment
  - Canted nozzle flange preferred
  - Alternate bolt-on adapter

INTERNAL INSULATION

- Material
  - Aft closure: Silica- and asbestos-filled NBR, e. g., Gen-Gard V-44 or equivalent
  - Segments and forward dome: Silica-filled NBR, e. g., Gen-Gard V-45 or equivalent
- Installation
  - Sheet stock layup
  - Cure in place
- Alternate being costed is mastic/cast insulation

PROPELLANT AND GRAINS

- Propellant type: PBAN (LPC-580 Modified)
- Propellant total weights
  - Parallel: 1.23×10^6 pounds
- Motor MEOP: 1000 psi
- Propellant characteristics
  - I_{sp} std: 262.6 seconds
  - Density: 0.0646 lb/in.^{3}
  - Burn rate at 1000 psi: 0.4 in./sec
- Grain configurations
  - Circular port or star as required

TOTAL INERT WEIGHT

- Parallel: 154×10^3 pounds

DOME SECTIONS (Common Fore and Aft)

- Material: D6AC
- Strength level: UTS 225 Ksi
- Biaxial gain: 13 percent
- Fabrication method
  - Roll formed, swaged/machined joints/no welds
  - Integral skirt, if possible with forging restrictions
  - Alternate: Bolt-on skirt
  - Two T/T ports canted 45 degrees

IGNITER

- Type: Pyrogen
- Case material: D6AC
- Propellant: LPC-580A
  - Grain weight: 500 pounds
  - Burn time: 0.5 second
- Insulation: asbestos-filled NBR
- Other
  - EBW type dual initiators

NOZZLE

- Submerged entrance (standard practice for movable nozzles)
- All ablative parts tape-wrapped
  - Carbon-phenolic
  - Silica-phenolic
  - Glass-phenolic
- Glass overwrap exit cone structure
- All steel parts - D6AC
- Inhibitors
  - As required
  - Silica-filled NBR

TVC

- N^2 cold-gas blowdown type system
- ±10 degree deflection
- 15 deg/sec slew rate
- Two hydraulic servo-actuators (90 degrees apart)
- Actuators are linear double-acting
- Design stall torque is 16×10^6 in. -lb
- System has capability for 20 full deflection cycles

The graphs in the following subsection show the results of tradeoffs of case length and mass fraction as a function of propellant weight and maximum expected operating pressure (MEOP). These results were obtained by varying the number of motor segments from 5 through 9 and by adhering to the following groundrules, which take into account the inputs received from the Phase B contractors:

Burn time (sec)	138
Nozzle exit diameter (in.)	154
Port-to-throat ratio	1.3:1

BASELINE PARALLEL BURN SRM WEIGHTS

• MOTOR CASE			
FWD SEGMENT	7,436		
CENTER SEGMENT 7 AT 11,500	80,500		
AFT SEGMENT	7,437		
TOTAL		95,373	
• INSULATION AND LINER			
FWD SEGMENT	1,986		
CENTER SEGMENT 7 AT 1,050	7,350		
AFT SEGMENT	4,635		
TOTAL		13,971	
• NOZZLE		17,004	
• THRUST TERMINATION		7,915	
• IGNITER		1,000	
• TVC AND LOCKSEAL		18,500	
TOTAL INERT			153,763
• PROPELLANT			
FWD SEGMENT	70,000		
CENTER SEGMENTS 6 AT 164,300	985,800		
CENTER SEGMENT STAR	135,000		
AFT SEGMENT	40,230		
TOTAL PROPELLANT			1,231,030
• TOTAL WEIGHT/MOTOR			1,384,793
• MASS FRACTION (WITH TVC AND TT)	.889		
• MASS FRACTION (WITHOUT TVC)	.901		
• MASS FRACTION (WITHOUT TVC AND TT)	.906		

SRM SAFETY FACTORS

1.4 MEOP ULTIMATE CASE STRENGTH

1.1 MEOP PROOF TEST ON CASE

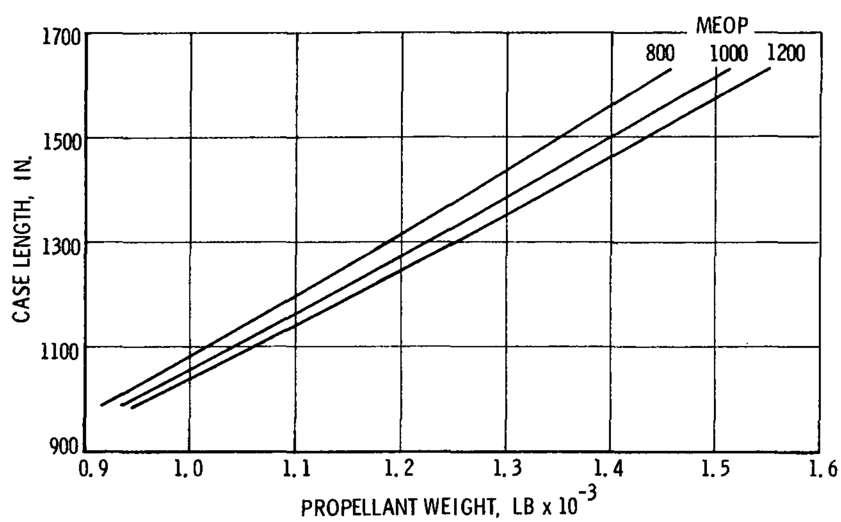
2.0 ON NOZZLE ABLATIVES

2.0 ON CASE INSULATION

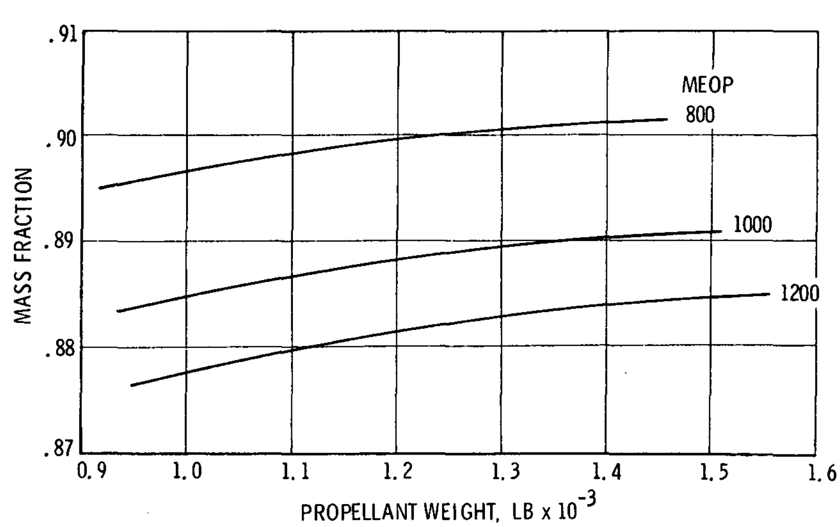
2.0 ON TVC PRESSURE TANKS, VALVES

2.5 ON TVC PLUMBING

CASE LENGTH vs PROPELLANT WEIGHT
PARALLEL BURN



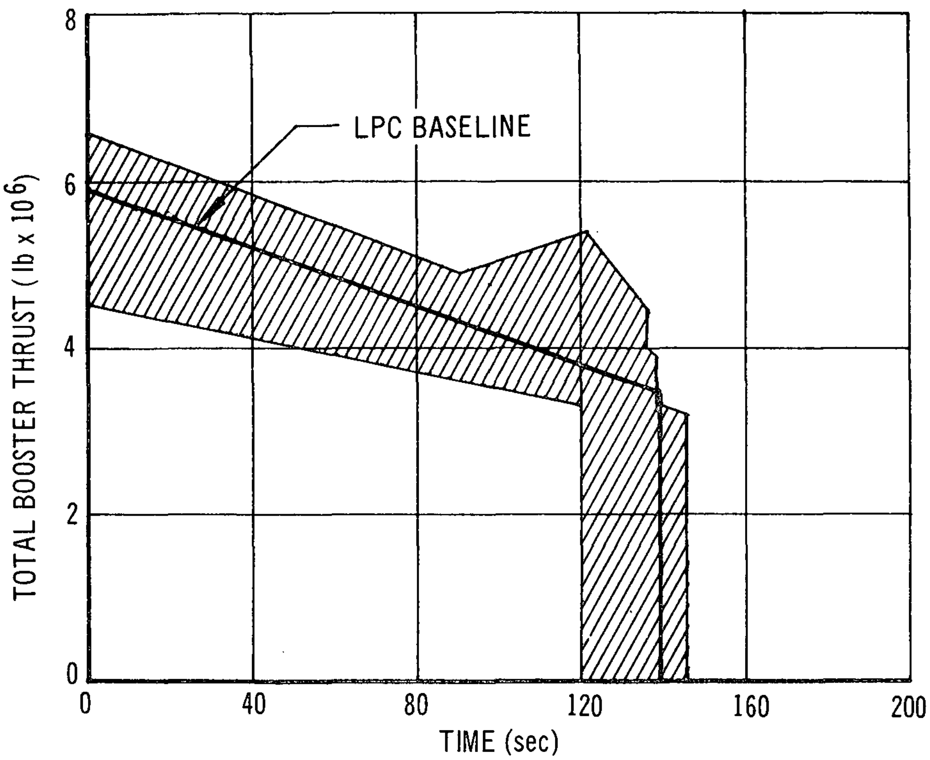
MASS FRACTION vs PROPELLANT WEIGHT
PARALLEL BURN



2.1.2 Performance

2.1.2.1 Ballistic Performance

Predicted thrust performance, together with selected motor characteristics for the LPC 156-7 baseline SRM design, is shown below. For reference, composite thrust-time requirements received from the Phase B prime contractors are depicted on the graph as a cross-hatched envelope.

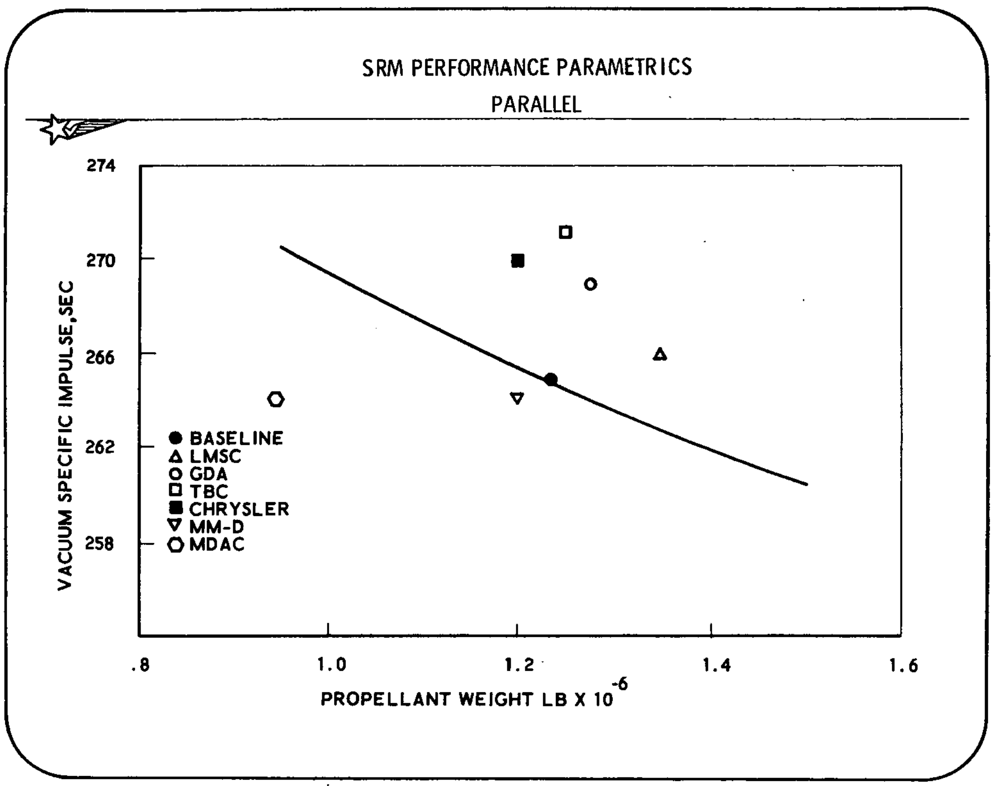


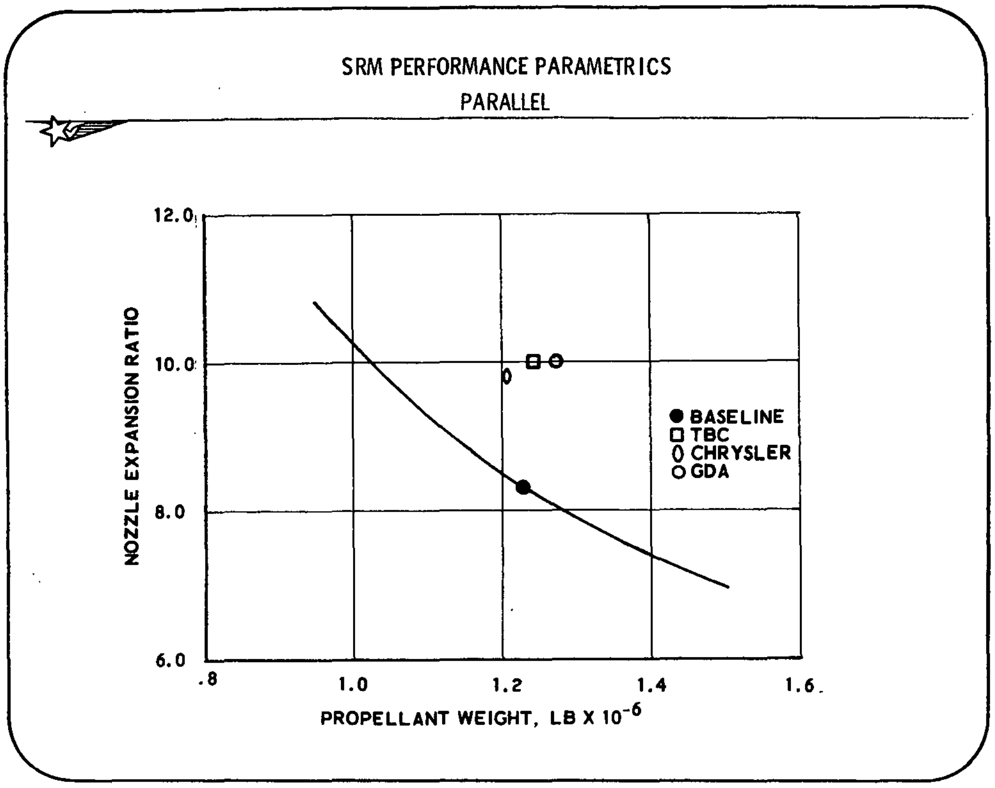
Parameter	Value
Initial thrust (lb x 10^6)	2.942
Burn time (sec)	138
Average pressure (psia)	631
MEOP (psia)	1000
Inert weight (lb)	153,763
Propellant weight (lb)	1,231,030
Mass ratio	0.889
Motor length (inches)	1494
I_{sp} vac del (sec) (initial)	264.8
Motor total weight (lb) (without TVC)	1,384,793
I_t vac (lbf-sec)	325,976,745

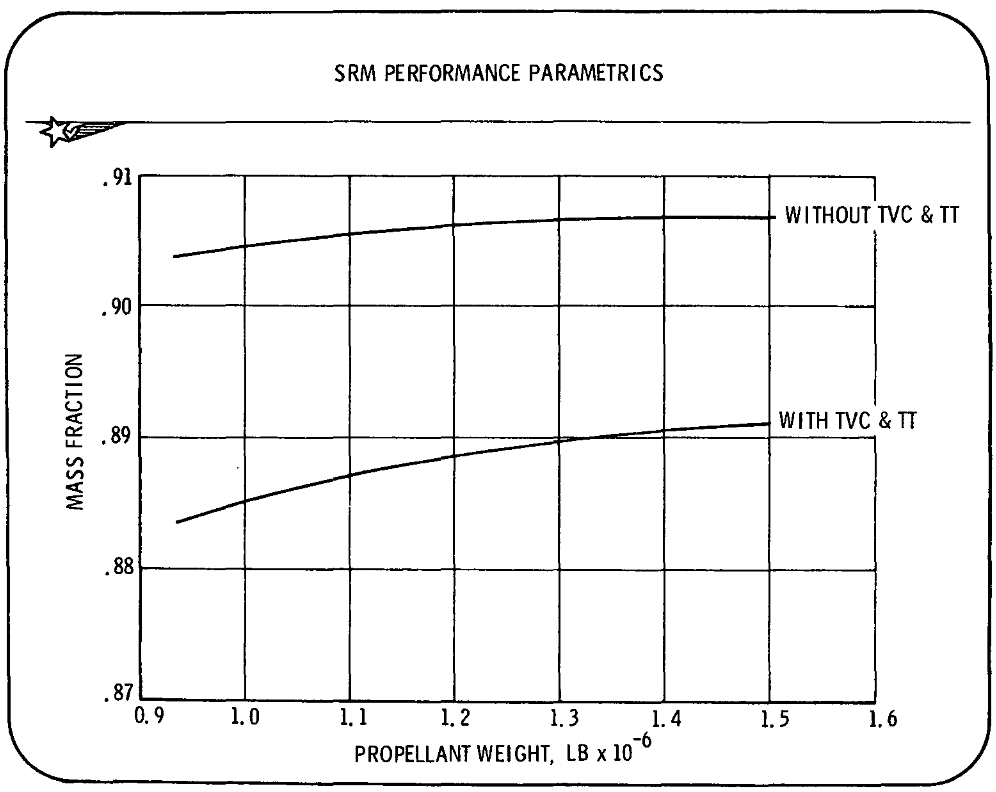
72-114

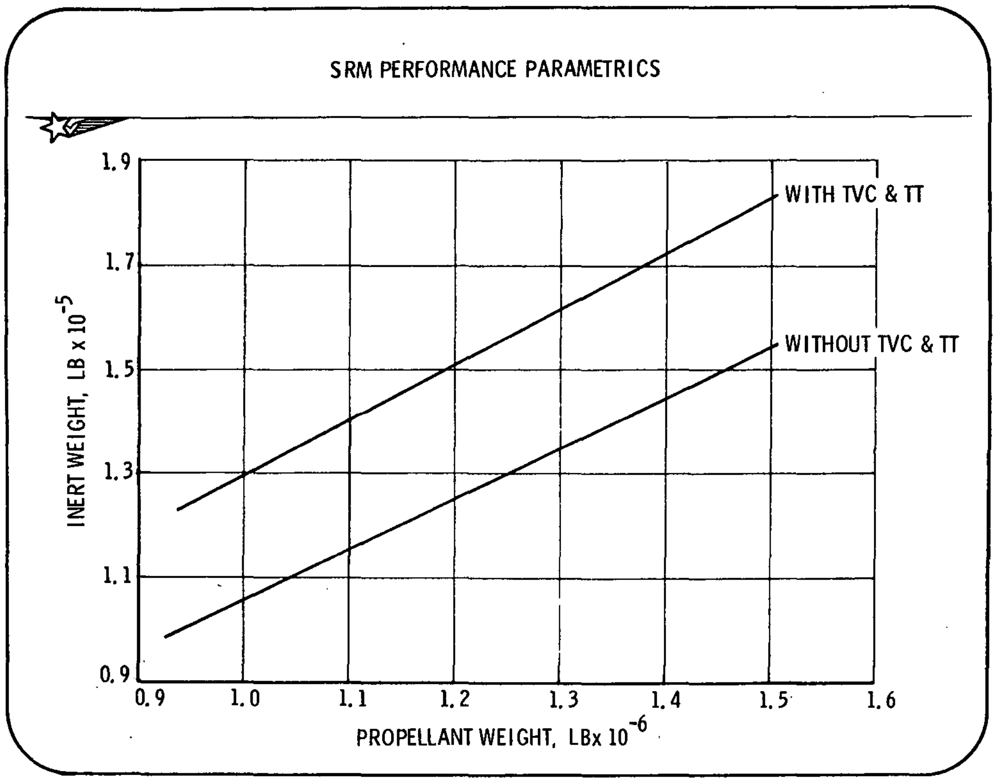
The following graphs show the results of tradeoffs (using the baseline propellant, case, and insulation materials) of vacuum specific impulse, expansion ratio, and SRM inert weight as a function of propellant weight. These results were obtained by varying the number of motor segments from 5 through 9 and by adhering to the following ground rules, which take into account the inputs received from the Phase B contractors:

Burn time (sec)	138
Maximum Expected Operating Pressure (psi)	1000
Case segment length (in.)	160
Nozzle exit diameter (in.)	154
Port-to-throat ratio	1.3:1




SRM PERFORMANCE PARAMETRICS
PARALLEL
NOZZLE EXPANSION RATIO
12.0
10.0
8.0
6.0
.8
1.0
1.2
1.4
1.6
PROPELLANT WEIGHT, LB X 10^-6
● BASELINE
□ TBC
0 CHRYSLER
O GDA

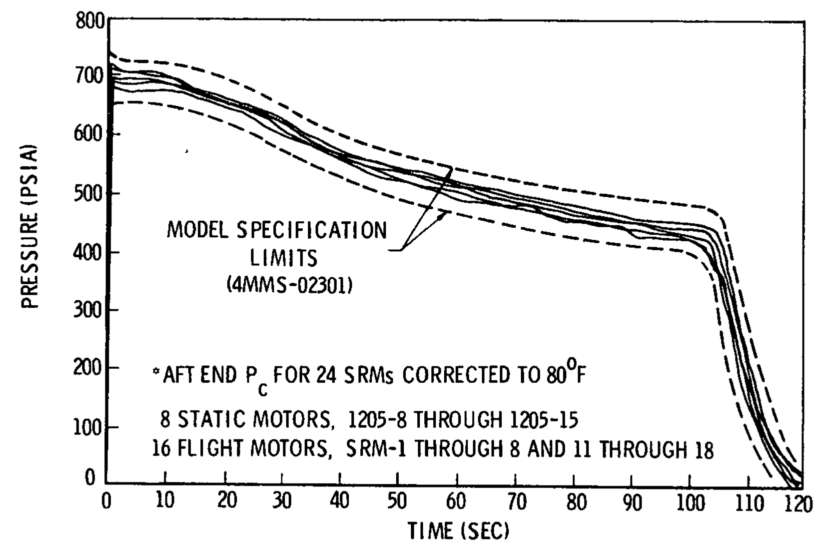

SRM PERFORMANCE PARAMETRICS
MASS FRACTION
.91
.90
.89
.88
.87
0.9
1.0
1.1
1.2
1.3
1.4
1.5
1.6
PROPELLANT WEIGHT, LB x 10^-6
WITHOUT TVC & TT
WITH TVC & TT


SRM PERFORMANCE PARAMETRICS
INERT WEIGHT, LB x 10^-5
1.9
1.7
1.5
1.3
1.1
0.9
0.9
1.0
1.1
1.2
1.3
1.4
1.5
1.6
PROPELLANT WEIGHT, LBx 10^-6
WITH TVC & TT
WITHOUT TVC & TT

Performance reproducibility. This subsection presents SRM performance reproducibility data based on static- and flight-test performance of twenty-four 120-inch SRM motors (the UA-1205 model used as zero stage in the Titan III vehicle). Shown below is the pressure-time performance for these motors with a superimposed model specification limits envelope. The computed three-standard-deviation statistics derived from these firings for several ballistic parameters of interest are also presented.

Finally, based on an analysis of the above tests, the expected thrust differential (between the two parallel burn baseline SRM's on a booster vehicle) as a function of time after web burnout is shown. The normalized data are presented as a two-standard deviation statistic.

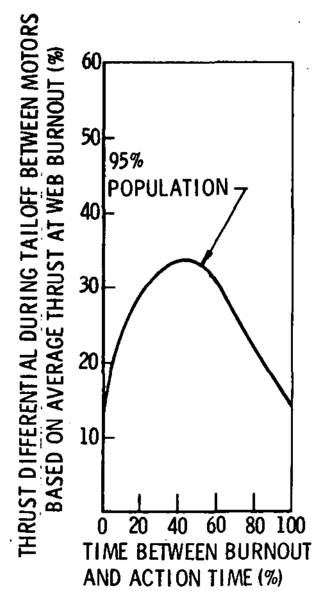
PRESSURE-TIME CURVE PERFORMANCE AND REPRODUCIBILITY
(UA-1205 SOLID ROCKET MOTORS*)



ABRIDGED UA 1205 PERFORMANCE SUMMARY
UNAUGMENTED NOZZLE CENTERLINE THRUST, 80°F

Parameter	Nominal	3-Sigma Limits (%)
Web time (sec)	104.1	±2.16
Action time (sec)	113.8	±3.43
Action time impulse (lbf sec x 10^{-6})	112.52	±1.0
Initial sea level thrust (lbf)	1,199,300	±6.23

156-INCH SRM PERFORMANCE REPRODUCIBILITY



2.1.2.2 Internal Acoustics

Some solid propellant motors generate sinusoidal vibrations resulting from acoustic pressure oscillations within the motor chamber. Such vibrations were measured on the Poseidon first- and second-stage motors[1] and on the third stage motor of Minuteman II and III[2]. On the other hand, they did not occur in Polaris motors, in other Minuteman stages, or in 156-inch motors previously tested. Considerable research has been devoted to this subject[3]; however, exact methods of predicting this phenomenon are not available. If these vibrations occur in the SRM, the most likely modes would be the longitudinal, closed-closed "organ pipe" modes, whose frequencies are predicted by the formula

$$f = \frac{Nv}{2l}$$

where

$N = 1, 2, 3, \ldots\ldots$

v = speed of sound in the chamber = 3550 ft/sec

l = length of the chamber

For the SRM, this gives 16 Hz, 32 Hz, etc.

With regard to Poseidon and Minuteman experience, the only problems were with electronic and hydraulic packages attached directly to the motor domes. These problems were solved by vibration isolation and minor packaging modifications. The largest measured pressures for the fundamental longitudinal mode were ±1.7 psi on the Poseidon motors. These very small pressures caused significant vibrations because the Poseidon and Minuteman motor domes are made of filament-wound glass, which is very flexible. The much stiffer (steel) motor domes of the SRM would have much lower acceleration responses.

---

(1) Pendleton, L.R., "Sinusoidal Vibration of Poseidon Solid Propellant Motors", presented at 42nd Shock and Vibration Symposium, Key West Florida, 4 November 1971

(2) Fowler, J.R., and Rosenthal, J.S., "Missile Vibration Environment for Solid Propellant Oscillatory Burning", presented at AIAA/SAE 7th Propulsion Joint Specialists Conference, Salt Lake City, Utah, 14-18 June 1971

(3) Culich, F.E.C., "Research on Combustion Instability and Application to Solid Propellant Rocket Motors', presented at AIAA/SAE 7th Propulsion Joint Specialist Conference, Salt Lake City, Utah, 14-18 June 1971

Motor-generated sinusoidal vibrations on Minuteman and Poseidon missiles caused no problems for missile structure or packages not attached directly on motor domes. In the analysis of available data on previous static-test firings of 156 inch motors containing the same propellant as planned for the SRM, no sinusoidal vibrations were observed.

2.1.3 Growth Potential

This subsection illustrates for the baseline design the growth potential that may be realized by adding one or two additional center segments and increasing the throat diameter. Growth is easily accomplished at low costs, as shown in the first two charts below, to provide flexibility to match program needs. The third chart indicates the relationship of chamber pressure design requirements to an increase in the number of motor segments. It is seen that case design pressure increases with the number of segments. This condition exists because of the decreasing port-to-throat area ratio (and a corresponding increase in head-end chamber pressure) resulting from a fixed port area and an increasing throat area requirement.

SRM BASELINE - GROWTH POTENTIAL

GROWTH READILY ACCOMPLISHED

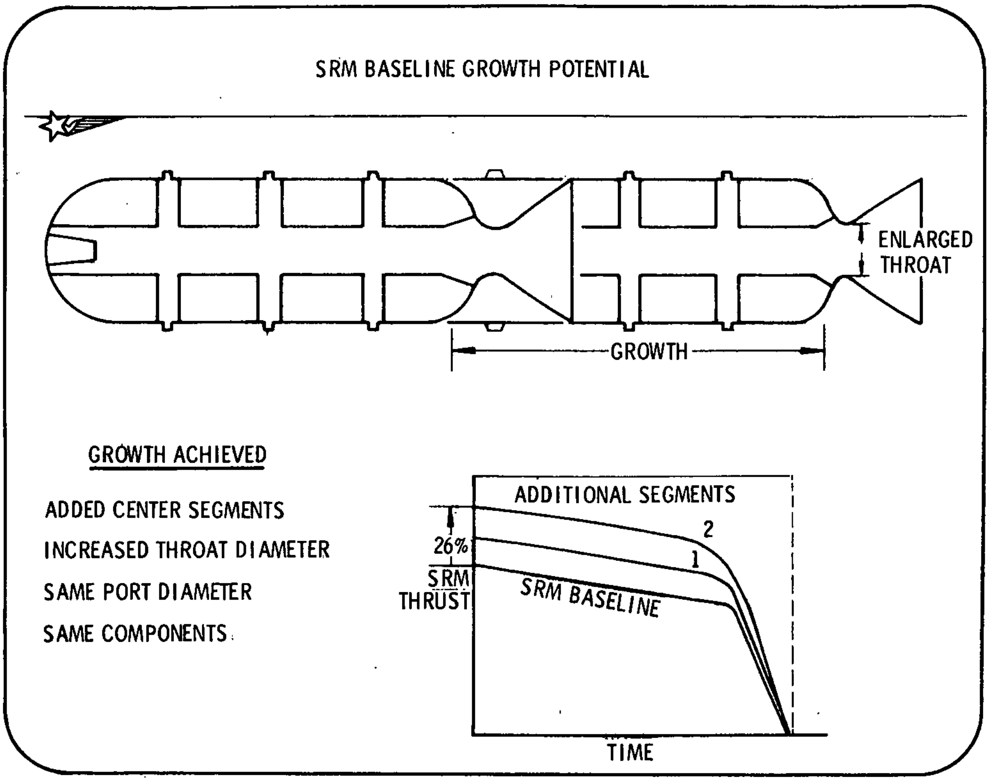
- INCREASE MOTOR LENGTH AND PROPELLANT
- SHORT TIME SPAN TO MAKE CHANGE IF NEEDED
- SIGNIFICANT PERFORMANCE GAIN AVAILABLE

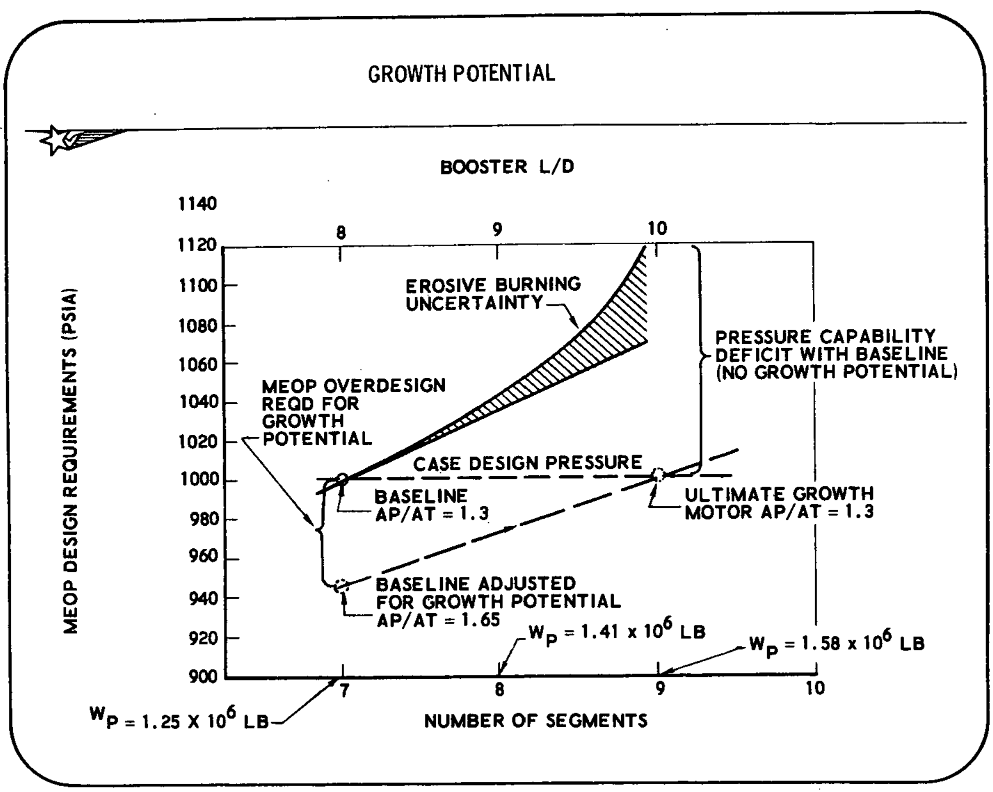
FLEXIBILITY TO SUIT PROGRAM NEEDS

- CAN PROVIDE FOR FUTURE GROWTH IN INITIAL DESIGN OF MOTOR (SMALL PENALTY)
- ALTERNATELY CAN EASILY MODIFY INITIAL DESIGN IF NOT PROVIDED FOR (SMALL DELAY)

COST TO PROVIDE IS SMALL

- ONE AND ONE-HALF PERCENT TO BOOSTER COST TO PROVIDE FOR INITIALLY
- MODIFY CASE MACHINING AND GRAIN TOOLING IF GROWTH NOT PROVIDED INITIALLY
- ONE DEVELOPMENT FIRING TO VERIFY MOTOR BALLISTICS AND NOZZLE INTEGRITY

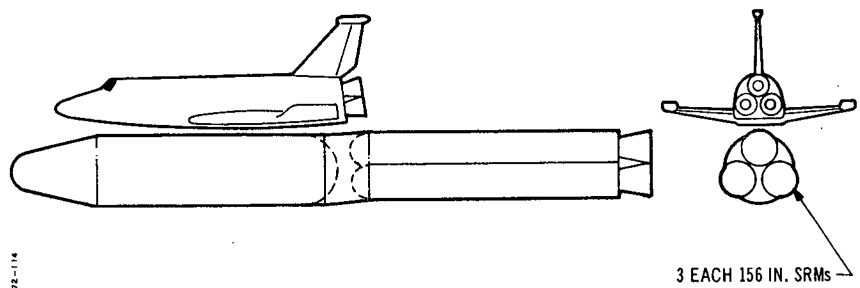

SRM BASELINE GROWTH POTENTIAL
ENLARGED THROAT
GROWTH
GROWTH ACHIEVED
ADDED CENTER SEGMENTS
INCREASED THROAT DIAMETER
SAME PORT DIAMETER
SAME COMPONENTS
ADDITIONAL SEGMENTS
2
1
26%
SRM THRUST
SRM BASELINE
TIME


GROWTH POTENTIAL
BOOSTER L/D
8
9
10
MEOP DESIGN REQUIREMENTS (PSIA)
1140
1120
1100
1080
1060
1040
1020
1000
980
960
940
920
900
EROSIVE BURNING UNCERTAINTY
PRESSURE CAPABILITY DEFICIT WITH BASELINE (NO GROWTH POTENTIAL)
MEOP OVERDESIGN REQD FOR GROWTH POTENTIAL
CASE DESIGN PRESSURE
BASELINE AP/AT = 1.3
ULTIMATE GROWTH MOTOR AP/AT = 1.3
BASELINE ADJUSTED FOR GROWTH POTENTIAL AP/AT = 1.65
W_P = 1.41 x 10^6 LB
W_P = 1.58 x 10^6 LB
W_P = 1.25 X 10^6 LB
7
8
9
10
NUMBER OF SEGMENTS

2.2 ALTERNATE SRM DESIGNS

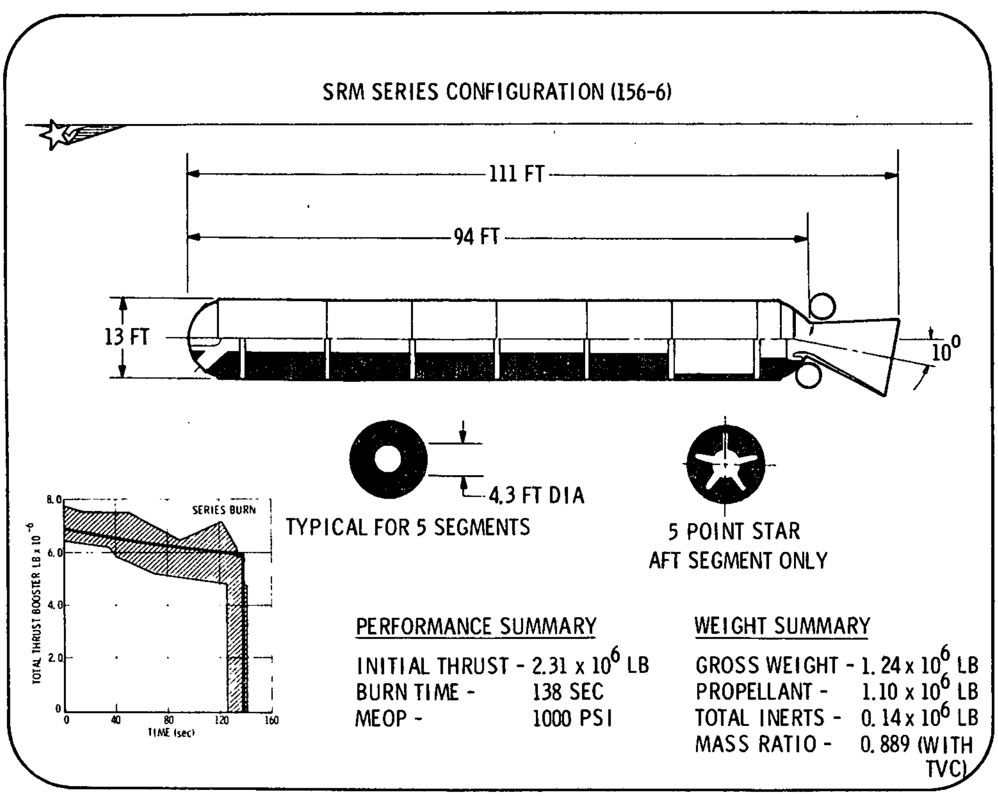
Data are presented in the following subsections for alternate SRM designs: a 156-inch series-burn configuration and a 120-inch configuration.

2.2.1 156-Inch Series-Burn SRM



	WEIGHT (LB X 10^{-6})
ORBITER LIFT-OFF WEIGHT	1.25
BOOSTER LIFT-OFF WEIGHT	3.82
GROSS LIFT-OFF WEIGHT	5.07

156-6 Series Boost System Space Shuttle



2.2.1.1 Design

SRM SERIES CONFIGURATION

CASE SEGMENTS

- Material: D6AC
- Strength level: UTS 225 Ksi
- Biaxial gain: 13 percent
- Fabrication method
  - Roll formed/machined joints/no welds
  - L/D = 1:1
  - Series: 6 Segments plus domes
- Joint configuration
  - Pin type: Tapered
  - Seal type: Barrel O-ring
- Nozzle attachment
  - Canted nozzle flange preferred
  - Alternate bolt-on adapter

INTERNAL INSULATION

- Material
  - Aft closure: Silica- and asbestos-filled NBR, e.g., Gen-Gard V-44 or equivalent
  - Segments and forward dome: Silica-filled NBR, e.g., Gen-Gard V-45 or equivalent
- Installation
  - Sheet stock layup
  - Cure in place
- Alternate being costed is mastic/cast insulation

PROPELLANT AND GRAINS

- Propellant type: PBAN (LPC-580 Modified)
- Propellant total weights
  - Series: 1.11×10^6 pounds
- Motor MEOP: 1000 psi
- Propellant characteristics
  - I_{sp} std: 262.6 seconds
  - Density: 0.0646 lb/in.^{3}
  - Burn rate at 1000 psi: 0.5 in./sec
- Grain configurations
  - Circular port or star as required

TOTAL INERT WEIGHT

- Series: 138×10^3 pounds

DOME SECTIONS (Common Fore and Aft)

- Material: D6AC
- Strength level: UTS 225
- Biaxial gain: 13 percent
- Fabrication method
  - Roll formed, swaged/machined joints/no welds
  - Integral skirt, if possible with forging restrictions
  - Alternate: Bolt-on skirt
  - Two T/T ports canted 45 degrees

IGNITER

- Type: Pyrogen
- Case material: D6AC
- Propellant: LPC-580A
  - Grain weight: 500 pounds
  - Burn time: 0.5 second
- Insulation: asbestos-filled NBR
- Other
  - EBW type dual initiators

NOZZLE

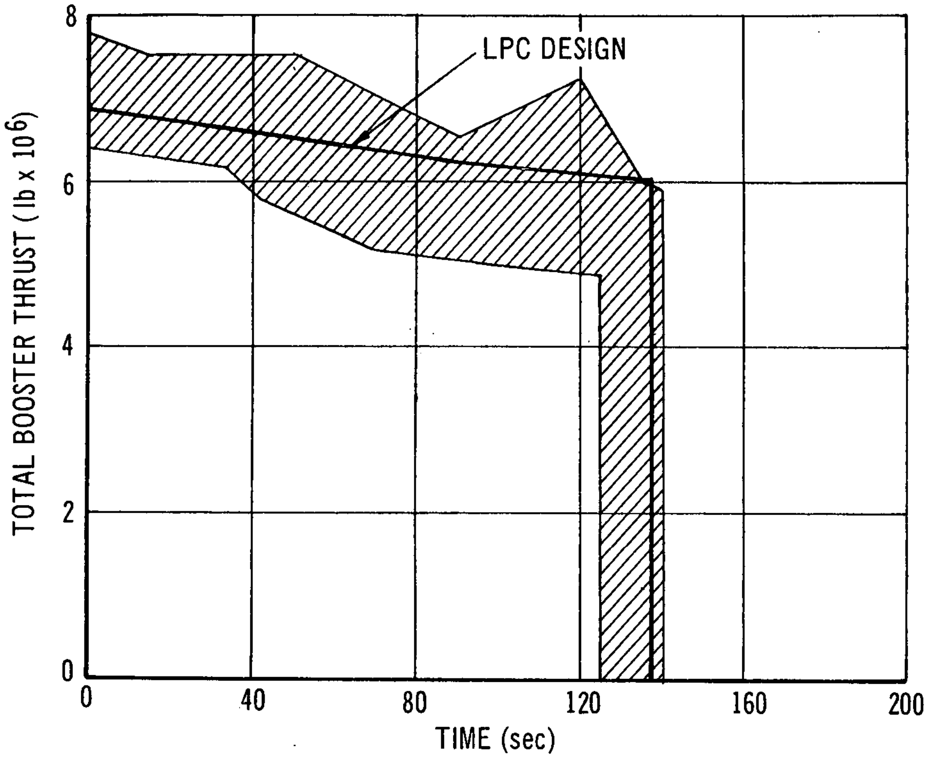
- Submerged entrance (standard practice for movable nozzles)
- All ablative parts tape-wrapped
  - Carbon-phenolic
  - Silica-phenolic
  - Glass-phenolic
- Glass overwrap exit cone structure
- All steel parts - D6AC
- Inhibitors
  - As required
  - Silica-filled NBR

TVC

- N^2 cold-gas blowdown type system
- ±10 degree deflection
- 15 deg/sec slew rate
- Two hydraulic servo-actuators (90 degrees apart)
- Actuators are linear double-acting
- Design stall torque is 16×10^6 in. -lb
- System has capability for 20 full deflection cycles

2.2.1.2 Performance

Predicted thrust performance, together with selected motor characteristics, for the LPC 156-6 series-burn SRM design is shown below. For reference, composite thrust-time requirements received from the Phase B prime contractors are depicted on the graph as a cross-hatched envelope.

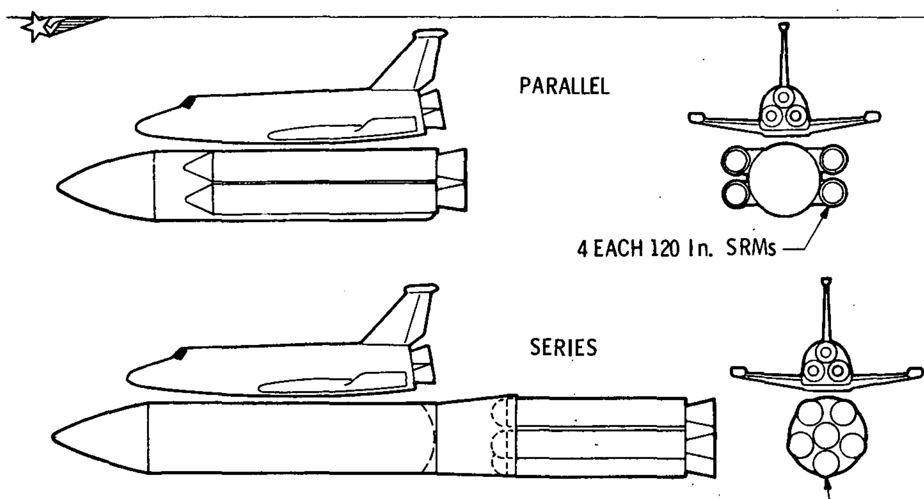


Parameter	Value
Initial thrust (lb x 10^6)	2.310
Burn time (sec)	138
Average pressure (psia)	741
MEOP (psia)	1000
Inert weight (lb)	137,750
Propellant weight (lb)	1,105,322
Mass ratio	0.880
Motor length (inches)	1334
I_{sp} vac del (sec) (initial)	270.6
Motor total weight (lb)	1,243,070
$I_{t\ vac}$ (lbf-sec)	299,099,592

2.2.2 120-Inch Motor

Presented first in this subsection are proposed configurations for Parallel-Burn and Series-Burn vehicles utilizing 120-inch SRMs. The following charts show design and performance details of two 120-7 SRM designs for both the Parallel and the Series Burn.

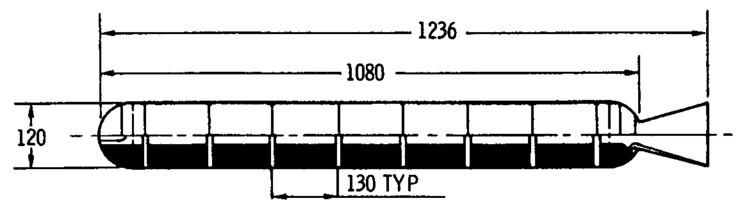
120-INCH VEHICLE BASELINE



WEIGHTS LB x 10^{-6}	PARALLEL	SERIES
OLOW	1.80	1.25
BLOW	2.85	4.28
GLOW	4.65	5.53

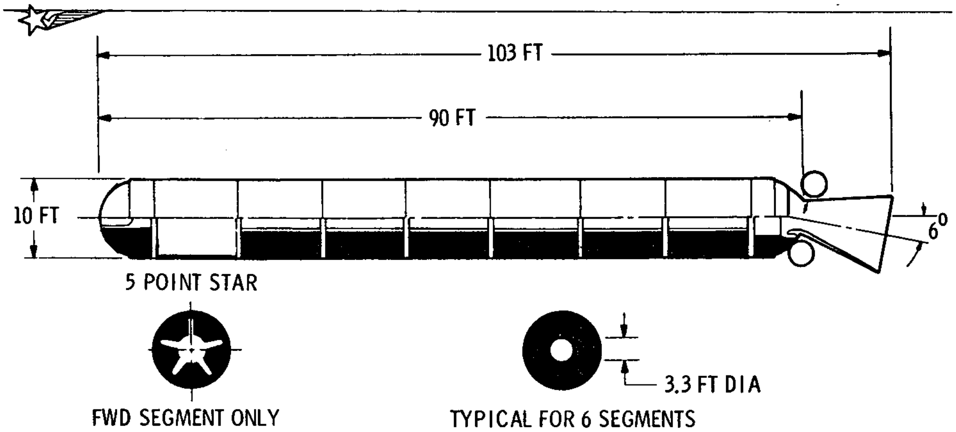
2.2.2.1 Design

**120-INCH VEHICLE CONFIGURATION
DESIGN FEATURES**



	PARALLEL	SERIES
BURN TIME (SEC)	140.7	144
INITIAL VACUUM THRUST, LB X 10^{-6}	1.40	1.59
TOTAL WEIGHT, LB X 10^{-6}	.697	.697
MASS RATIO	.85	.85

SRM PARALLEL CONFIGURATION (120-7)



PERFORMANCE SUMMARY

INITIAL THRUST - 1.4 x 10^6 LB
BURN TIME - 140 SEC
MEOP - 920 PSI

WEIGHT SUMMARY

GROSS WEIGHT - 0.697 x 10^6 LB
PROPELLANT - 0.593 x 10^6 LB
TOTAL INERTS - 0.104 x 10^6 LB
MASS RATIO - 0.85
CASE 48,000
NOZZLE 9,500
TVC 15,000

REF: UTC 2401-ISR-1, 5-20-71

2.2.2.2 Performance

DESIGN AND PERFORMANCE DETAILS FOR 120-INCH MOTOR

(Same as UTC 1207)

SERIES BURN

Design Characteristics	
Action time (sec)	144.4
Total impulse (vac)(lbf-sec)	1.59×10^8
Initial thrust (SL) (lbf)	1.20×10^6
Throat area (in.^{2})	1,359
Expansion ratio	9.2
Chamber pressure (max) (psia)	800
Mass fraction (motor only)	0.890
Case material	D6AC
Nozzle cant (deg)	None
Nozzle slew rate (deg/sec)	5
Propellant	
Specific impulse (vac) (sec)	268.7
Characteristic velocity (ft/sec)	5,170
Density (lbm/in.^{3})	0.063
Binder type	PBAN
Weights (lb)	
Propellant	592,000
Case	48,000
Nozzle	7,000
Other inert	18,000
Total motor	665,000

PARALLEL BURN

Design Characteristics	
Action time (sec)	140.7
Total impulse (vac) (lbf-sec)	1.59×10^8
Initial thrust (SL) (lbf)	1.40×10^6
Throat area (in.^{2})	1,359
Expansion ratio	9.2
Chamber pressure (max) (psia)	800
Mass fraction (motor only)	0.890
Case material	D6AC
Nozzle cant (deg)	None
Nozzle slew rate (deg/sec)	5
Propellant	
Specific impulse (vac) (sec)	268.7
Characteristic velocity (ft/sec)	5,170
Density (lbm/in ^{3})	0.063
Binder type	PBAN
Weights (lb)	
Propellant	592,000
Case	48,000
Nozzle	7,000
Other inert	18,000
Total motor	665,000

Section 3

SUBSYSTEM DEFINITION

This section contains a definition of motor components including case, nozzle, igniter, internal insulation, propellant, thrust termination system, and thrust vector control system. It also includes the results of subsystem safety/hazards analysis and man-rating and reliability studies. Drawings, descriptions, and applicable interface requirements are presented.

The selected features and materials for each baseline SRM component are enumerated, together with alternatives and corresponding selection rationale, and summarized on the following pages. The primary basis for selection was demonstrated experience. This approach provides for a minimum-risk booster development program and the availability of cost information based on actual experience. Each of the chosen component approaches has an extensive production history.

SRM COMPONENTS

WHAT

1.	MOTOR CASE	D6AC, 225 KSI ULTIMATE 160-INCH SEGMENT ROLL FORMED / NO WELDS
2.	NOZZLE	ABLATIVE PLASTIC THROAT D_t = 52.3 IN., ϵ = 8.33
3.	IGNITER	HEAD END PYROGEN, PBAN PROPELLANT B-KNO_3 PELLETS, EBW INITIATOR
4.	INTERNAL INSULATION	FILLED NBR CALENDERED SHEET STOCK, AUTOCLAVE CURE
5.	PROPELLANT	PBAN, LPC-580 87% SOLIDS, 18% ALUMINUM

SRM COMPONENTS

WHAT

6. THRUST TERMINATION
   - USED ON ABORT ONLY
   - DUAL HEAD END PORTS
   - SHAPED CHARGE COVER REMOVAL
   - REDUNDANT INITIATION SYSTEM

7. THRUST VECTOR CONTROL
   - SYSTEM CAPABILITY*
     - ± 10 DEGREES DEFLECTION ANGLE (OMNIAXIAL)
     - 15 DEGREES / SECOND SLEW RATE
     - 25 PERCENT DUTY CYCLE (21 FULL DEFLECTIONS)
   - SYSTEM FEATURES
     - LOCKSEAL FLEXIBLE JOINT
     - HYDRAULIC ACTUATORS
     - COLD GAS BLOWDOWN POWER SUPPLY

* VERY CONSERVATIVE REQUIREMENTS

SELECTION RATIONALE
SRM COMPONENTS

WHY	COSTS
1. EXTENSIVE PRODUCTION ON MINUTEMAN, TITAN III, SRAM	1. D6AC SAVES $ 140M COMPARED TO MARAGING
2. LOW RISK; MATERIALS PROVEN 156-IN., 260-IN. AND TITAN III	2. POTENTIAL SAVINGS OF $89M WITH ALTERNATE MATERIALS
3. HEAD END PYROGENS DEMONSTRATED ON LPC 156-INCH SRM's AND TITAN III	3. MINIMUM DEVELOPMENT COSTS
4. PROVEN COMPATIBILITY WITH PBAN PROPELLANT. PROVEN RELIABILITY.	4. MINIMAL PROCESS/MATERIAL COSTS
5. FULLY DEMONSTRATED ON MINUTEMAN, TITAN III, AND PRIOR 156 and 260-INCH SRM's.	5. RAW MATERIAL COST S: PBAN = $0.263 / LB; HTPB CANDIDATE = $0.255 / LB

SELECTION RATIONALE (continued)
SRM COMPONENTS

WHY

6. BASIC APPROACH SUCCESSFULLY DEMONSTRATED ON POSEIDON, MINUTEMAN AND TITAN III OVER 12-YEAR PERIOD

POSEIDON	-	42 FLIGHTS (IMPULSE CONTROL)
MINUTEMAN	-	150 FLIGHTS (IMPULSE CONTROL)
120-INCH SRM	-	1 GROUND TEST

EXHAUST PLUME AND DEBRIS WILL NOT DAMAGE ORBITER OR TANKS

THERMAL INPUT LOW - ONLY 12 BTU/FT^2-SEC

IMPINGEMENT PRESSURE LOW - ONLY 2 PSIA

MINIMUM DEBRIS CLEARANCE - 7.5 FEET FOR H/O TANK
17.5 FEET FOR ORBITER

7. LOCKSEAL HAS 100 FLIGHTS WITH TOTAL SUCCESS (USED ON POSEIDON)

SATURN IC ACTUATORS WITH MODIFICATIONS

COLD GAS SYSTEM IS SIMPLE AND RELIABLE (USED ON TITAN III)

3.1 MOTOR CASE

Only materials and processes with a background of successful experience in similar applications were considered. The candidate materials included D6AC, 18-percent nickel maraging, and HY-140 steels. The segment length (and thus the number of segments per motor) is dependent upon the material and the fabrication method used. Cylinder segments of D6AC made by roll-forming are limited to approximately 160 inches in length unless advancements are made in the current state-of-the-art. Larger segments can be obtained by welding two or more formed cylinders together, or by rolling and welding the entire cylinder. Such method would require advancements in welding technology.

The selected motor case design utilizes D6AC steel, with 160-inch long, roll-formed cylinder segments. This fabrication method has been successfully demonstrated on the Titan III motor. It provides reliability with low risk and at low cost. Development costs are especially low with this approach because technology advancement is not required.

MOTOR CASE ALTERNATES

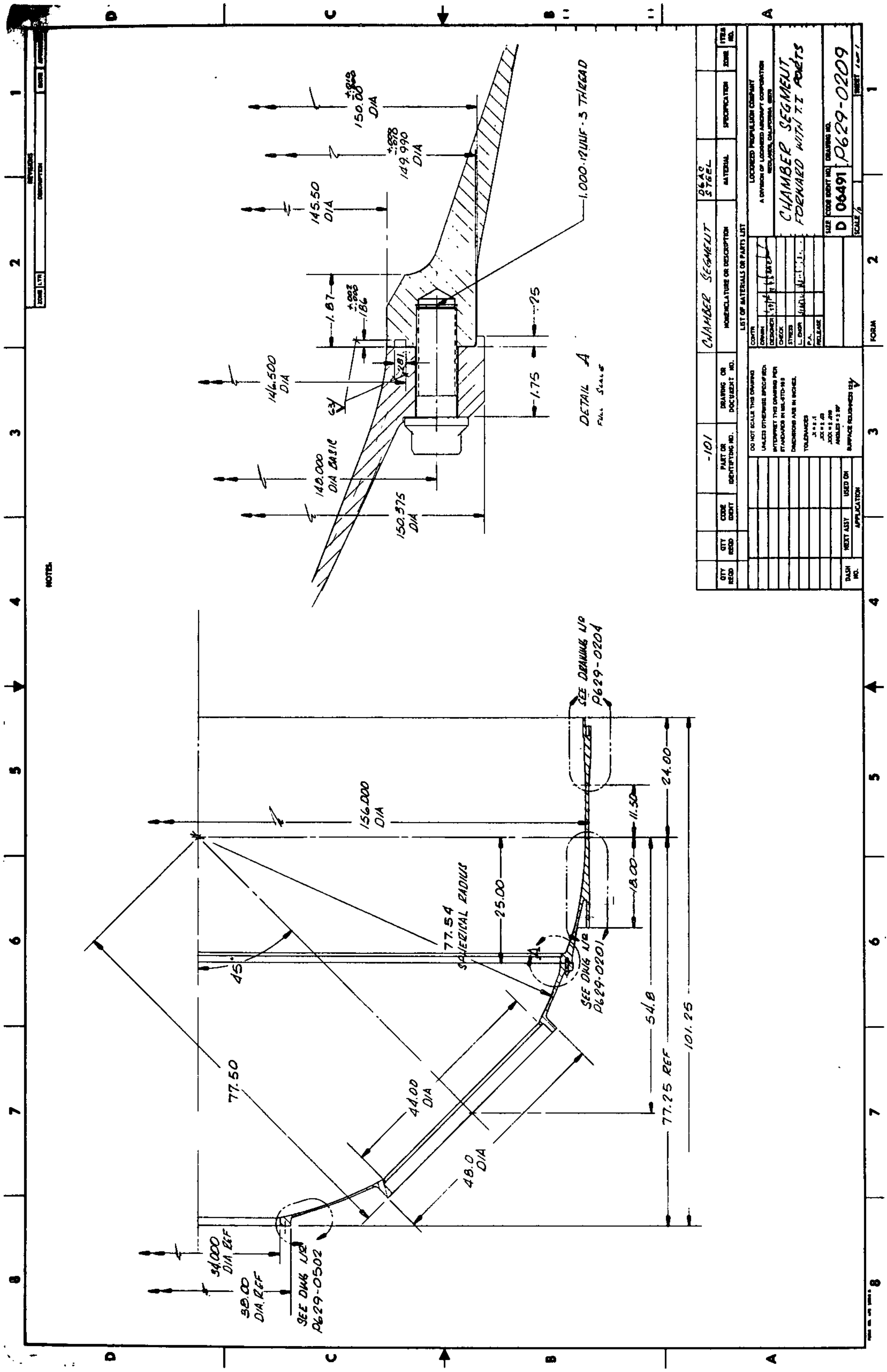
APPROACH	TECHNOLOGY ADVANCEMENT ALTERNATES	BURST TESTS	CORROSION SYSTEM	METALLUREY	HT FACILITY	WELD INSPECTION	WELD TECHNIQUE	ROLL FORM TECHNIQUE	MACHINING TECHNIQUES
D6AC STEEL 160 IN. SEGMENTS		◆	◆	◆					
D6AC STEEL 230 IN. SEGMENTS	LARGE BILLET TECHNOLOGY ROLL FORMING 1 : 1 RATIO	◆	◆	◆	◆			◆	
D6AC STEEL 230 IN. SEGMENTS WITH WELD	WELD TECHNIQUES	◆	◆	◆	◆	◆	◆		
MARAGING STEEL 160 IN. SEGMENTS	ROLL FORM TECHNIQUES	◆	◆	◆				◆	◆
MARAGING STEEL SEGMENTS 230 IN. WITH ROLL AND WELD	WELD TECHNIQUE FOR GIRTH AND LONG. WELDS	◆	◆	◆		◆	◆		◆
HY-140 STEEL	WELD TECHNOLOGY	◆	◆	◆		◆	◆		◆

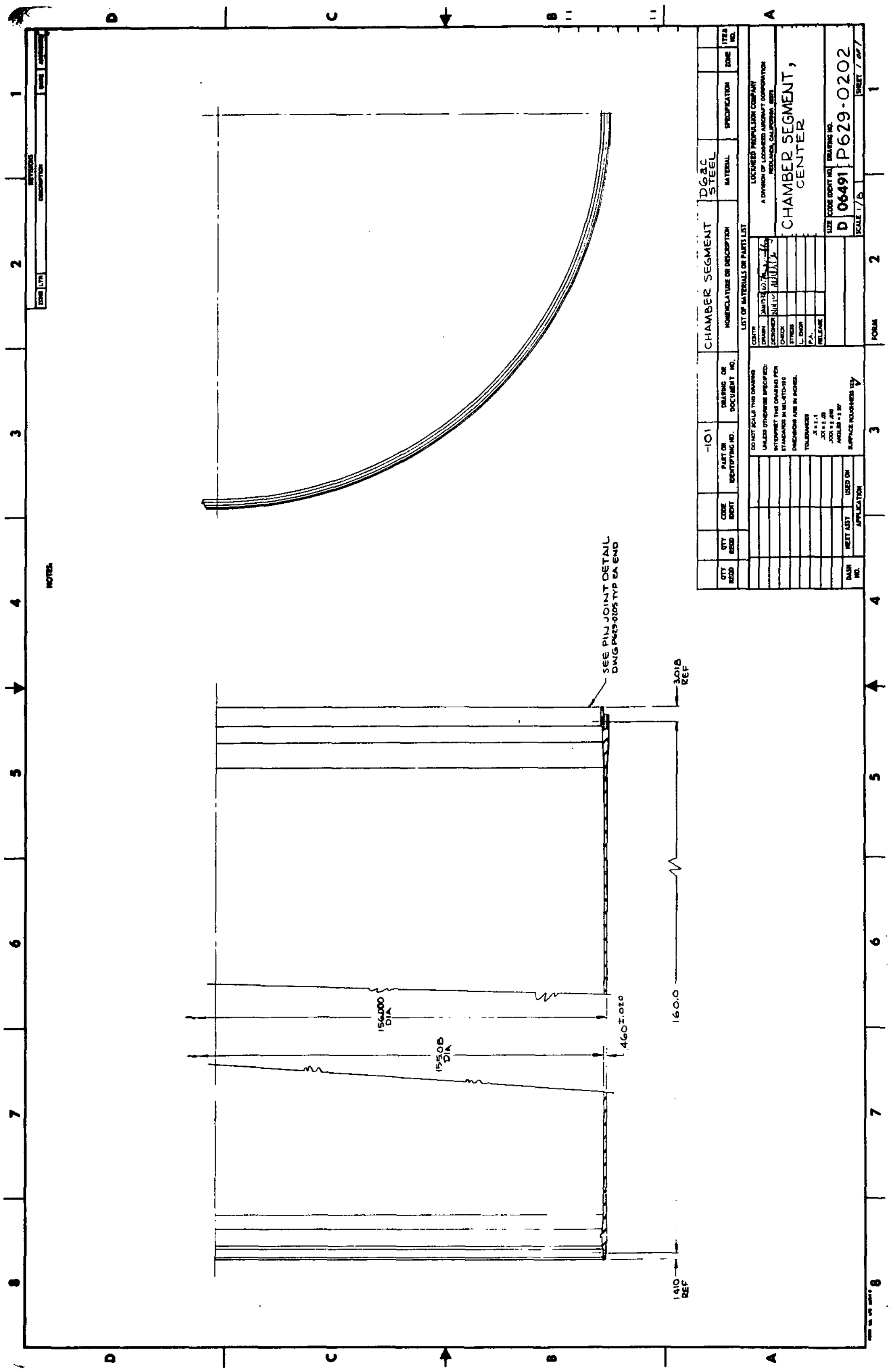
TRADE-OFF OF CASE FABRICATION APPROACHES

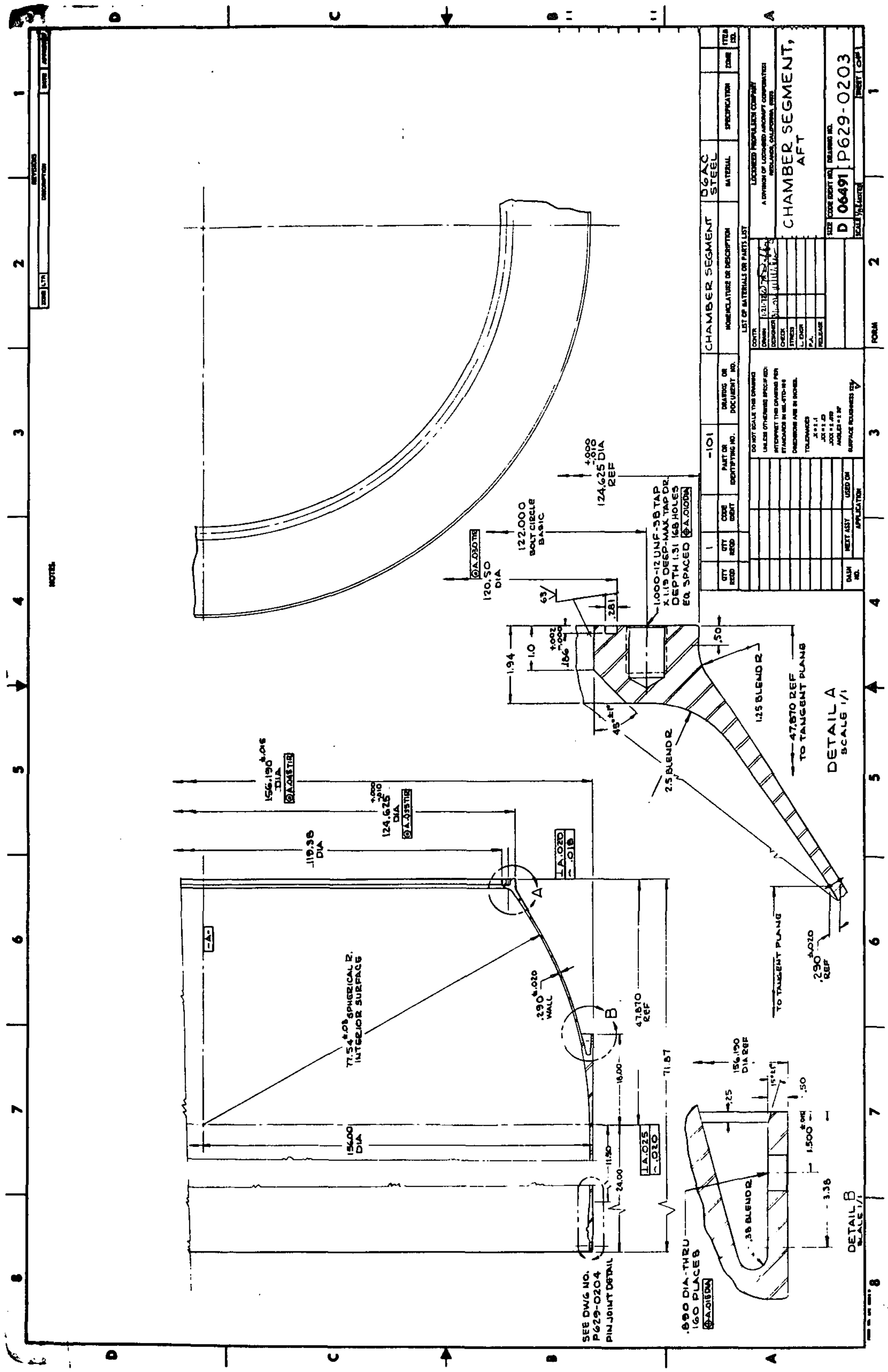
STEEL CASE	F_{TU} KSI	SEGMENT FABRICATION	NO. OF SEGMENTS SERIES	NO. OF SEGMENTS PARALLEL	SEGMENT LENGTH (IN.)
D6AC	225	RING ROLL, ROLL FORM, HEAT TREAT	6	7	160
D6AC	225	RING ROLL, ROLL FORM, HEAT TREAT	4	5	230
D6AC	225	RING ROLL, ROLL FORM, GIRTH WELD, HEAT TREAT	4	5	230
18% Ni MARAGING	225	RING ROLL, ROLL FORM, AGE	7	8	140
18% Ni MARAGING	225	ROLL AND WELD, AGE	4	5	230
HY-140	150	HEAT TREAT, ROLL AND WELD	4	5	230

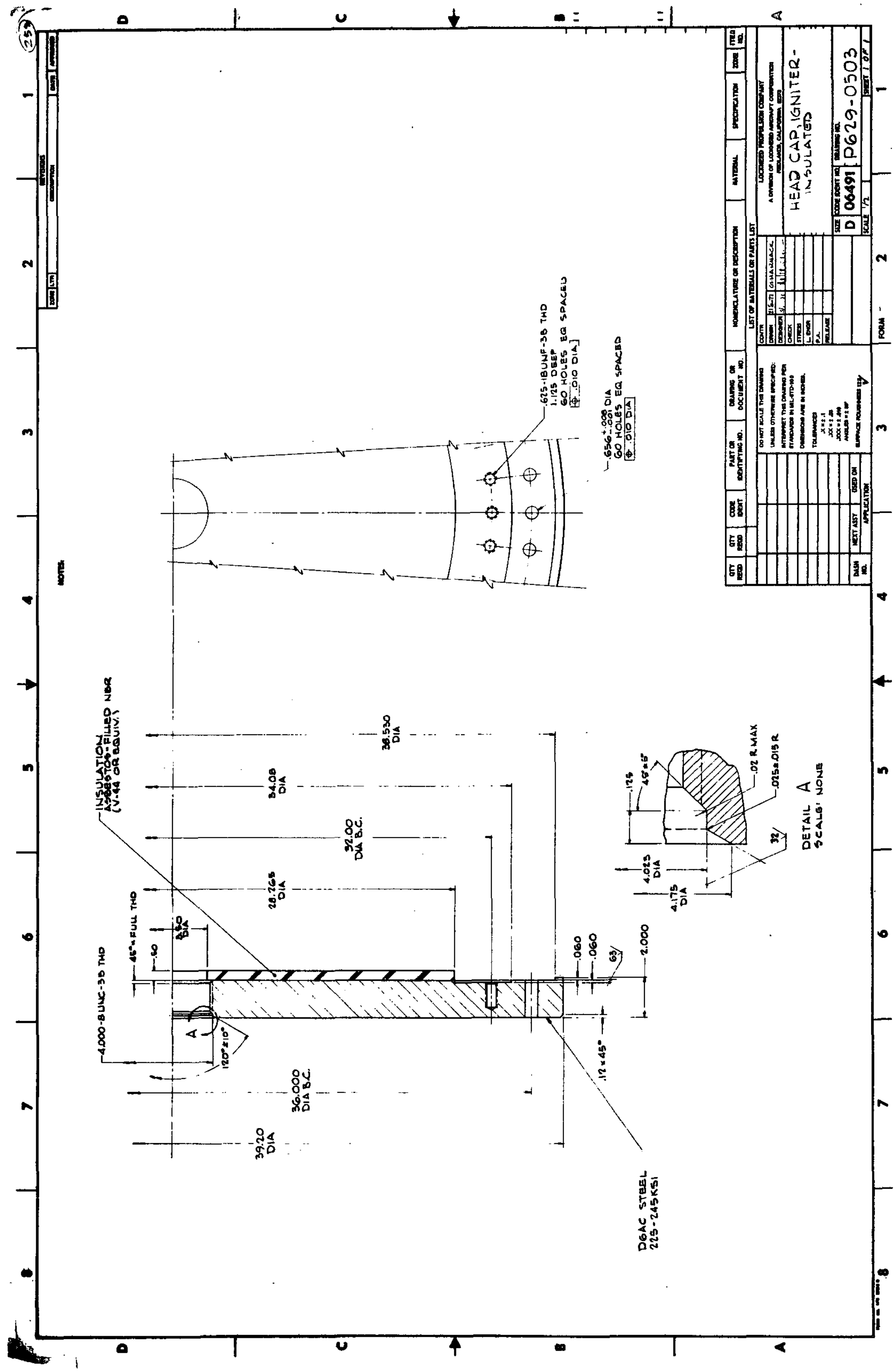
SELECTED CASE FABRICATION

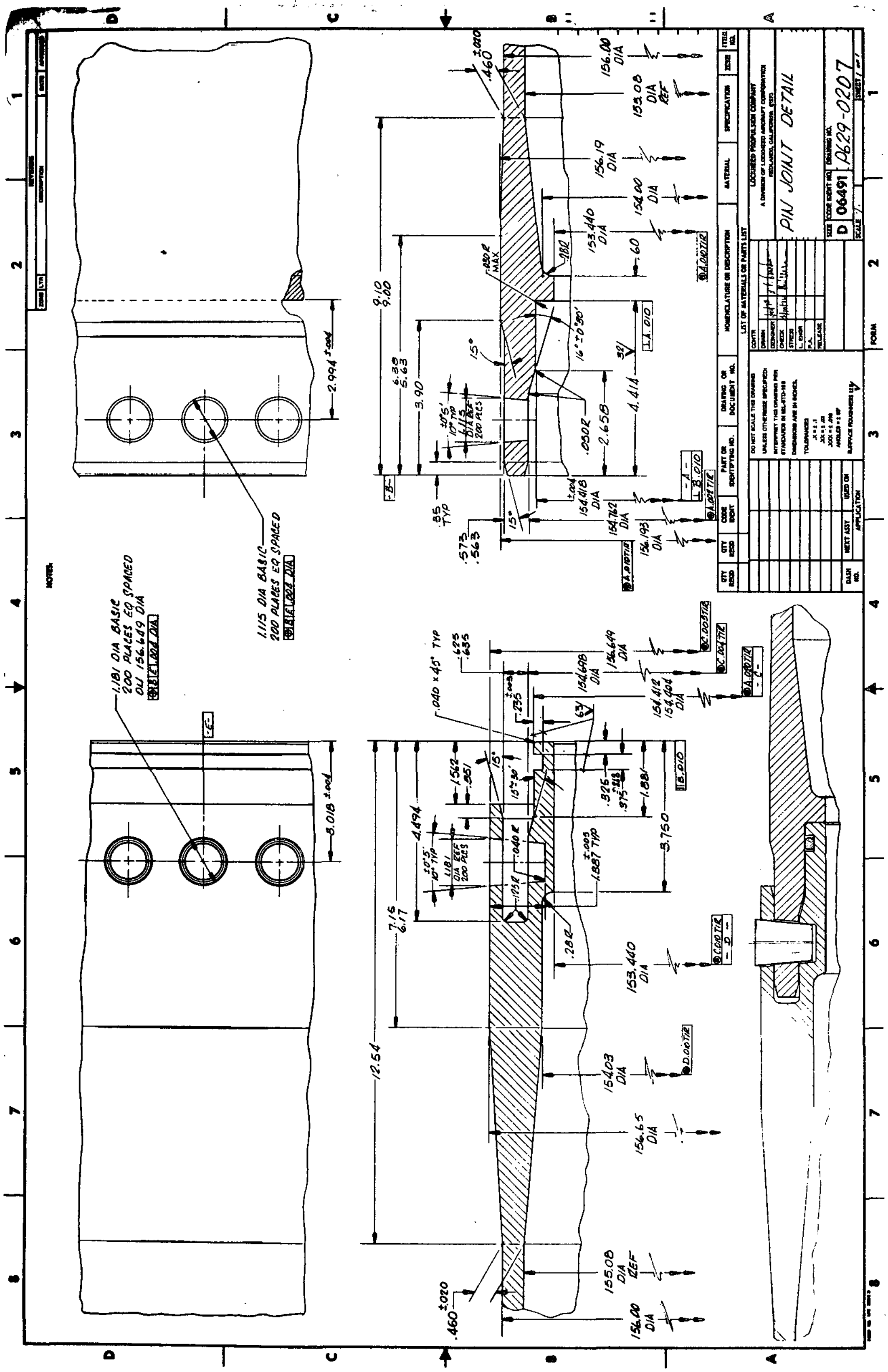
FEATURE	SELECTION	RATIONALE
CASE MATERIAL	D6AC STEEL	EXPERIENCE
FABRICATION METHOD	160-INCH SEGMENTS - RING ROLL, ROLL FORM, HEAT TREAT CLOSURES - RING ROLL, SWAGE, HEAT TREAT	RELIABILITY WITH NO WELDS. EXPERIENCE, FIRM COSTS
JOINT DESIGN	CLEVIS TYPE WITH TAPERED PINS BOLTED ON SKIRTS	STIFFNESS AND REUSABILITY
WALL THICKNESS	0.46-INCH NOMINAL FOR 1000 PSI MEOP	EXPERIENCE 1.4 SAFETY FACTOR 13% BIAXIAL GAIN 225 KSI F_{TU} MINIMUM
CORROSION PREVENTIVE	PAINT PLUS SEALANT SYSTEM	EXPERIENCE


CHAMBER SEGMENT FORWARD WITH T.I. PORTS
D 06491 P629-0209
CHAMBER SEGMENT
DETAIL A
1.000-12UNF-3 THREAD
SEE DWG NO P629-0201
SEE DRAWING NO P629-0204
SEE DWG NO P629-0502
77.54 SPHERICAL RADIUS
156.000 DIA
44.00 DIA
48.0 DIA
101.25
77.25 REF
77.50
LOCKHEED PROPULSION COMPANY


CHAMBER SEGMENT
D6AC STEEL
CHAMBER SEGMENT, CENTER
D 06491 P629-0202
SEE PIN JOINT DETAIL
160.0


CHAMBER SEGMENT, AFT
D 06491 P629-0203
CHAMBER SEGMENT
D6AC STEEL
LOCKHEED PROPULSION COMPANY
DETAIL A
SCALE 1/1
DETAIL B
SCALE 1/1
SEE DWG NO. P629-0204 PIN JOINT DETAIL
.890 DIA-THRU 160 PLACES
1.000-12 UNF-3B TAP X 1.19 DEEP-MAX TAP DR. DEPTH 1.51 168 HOLES EQ. SPACED
122.000 BOLT CIRCLE BASIC
77.54 SPHERICAL R. INTERIOR SURFACE
NOTES:


HEAD CAP, IGNITER-INSULATED
D 06491 P629-0503
LOCKHEED PROPULSION COMPANY
INSULATION ASBESTOS-FILLED NBR (V-44 OR EQUIV.)
D6AC STEEL 225-245 KSI
DETAIL A SCALE: NONE
.625-18UNF-3B THD 1.125 DEEP 60 HOLES EQ SPACED
.656+.005 -.001 DIA 60 HOLES EQ SPACED
4.000-8UNC-3B THD
36.000 DIA B.C.
39.20 DIA
38.530 DIA
34.08 DIA
32.00 DIA B.C.
28.765 DIA
2.000
NOTES:


PIN JOINT DETAIL
D 06491 D629-0207
LOCKHEED PROPULSION COMPANY

3.2 NOZZLE

Nozzle tradeoff studies included the examination of three primary design approaches for the baseline nozzle configuration:

- Fully qualified state-of-the-art materials
- Advanced state-of-the-art, moderate-cost materials
- Advanced state-of-the-art, low-cost materials

Program nozzle cost studies were conducted for both the fully qualified, state-of-the-art design and the advanced state-of-the-art, low-cost design.

The low risk state-of-the-art design was chosen. It incorporates fully proven configurations and materials at relatively low program costs.

Details of the tradeoffs and the selected design are presented on the following pages.

NOZZLE MATERIALS CONSIDERED

MATERIALS ALTERNATIVE	COMPONENT: NOSE CAP AND ENTRANCE	THROAT INSERT	FORWARD EXIT CONE	AFT EXIT CONE	LOCKSEAL HEAT BARRIER
STATE OF THE ART APPROACH					
CARBON/PHENOLIC	●	●	●		
SILICA/PHENOLIC				●	●
MODERATE COST DESIGN					
CARBON/PHENOLIC		●			
LOW COST CARBON/PHENOLIC	●		●		
LOW COST SILICA/PHENOLIC				●	●
LOW COST DESIGN					
CARBON/PHENOLIC		●			
LOW COST CARBON/PHENOLIC	●		●		
CANVAS/PHENOLIC				●	●

NOZZLE FABRICATION APPROACHES

CONFIGURATION: BASED ON STATE OF THE ART DESIGN STANDARDS

FABRICATION METHODS: ALL PARTS TAPE WRAPPED

MATERIALS: ABLATIVE

	MAXIMUM USE OF:
STATE OF THE ART, FULLY QUALIFIED MATERIALS	➧ CARBON CLOTH/PHENOLIC SILICA CLOTH/PHENOLIC
ADVANCED STATE OF THE ART, MODERATE COST MATERIALS	➧ LOW COST CARBON CLOTH/ PHENOLIC LOW COST SILICA CLOTH/ PHENOLIC
ADVANCED STATE OF THE ART, LOW COST MATERIALS	➧ CANVAS DUCK/PHENOLIC LOW COST CARBON CLOTH/ PHENOLIC LOW COST SILICA CLOTH/ PHENOLIC

SRM COMPONENTS
NOZZLE

WHAT

D6AC STEEL STRUCTURAL METAL PARTS

ABLATIVE PLASTIC THROAT

THROAT DIAMETER 52.3 INCHES

EXPANSION RATIO 8.3

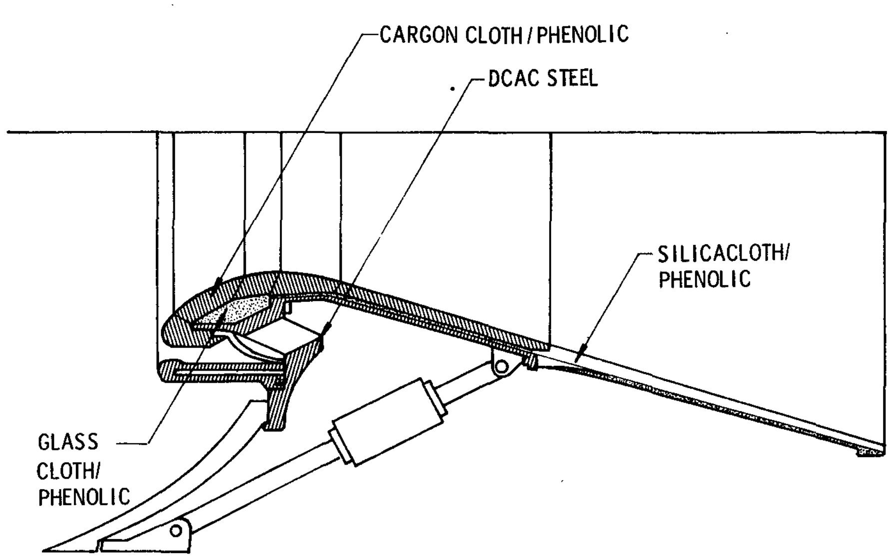
WHY

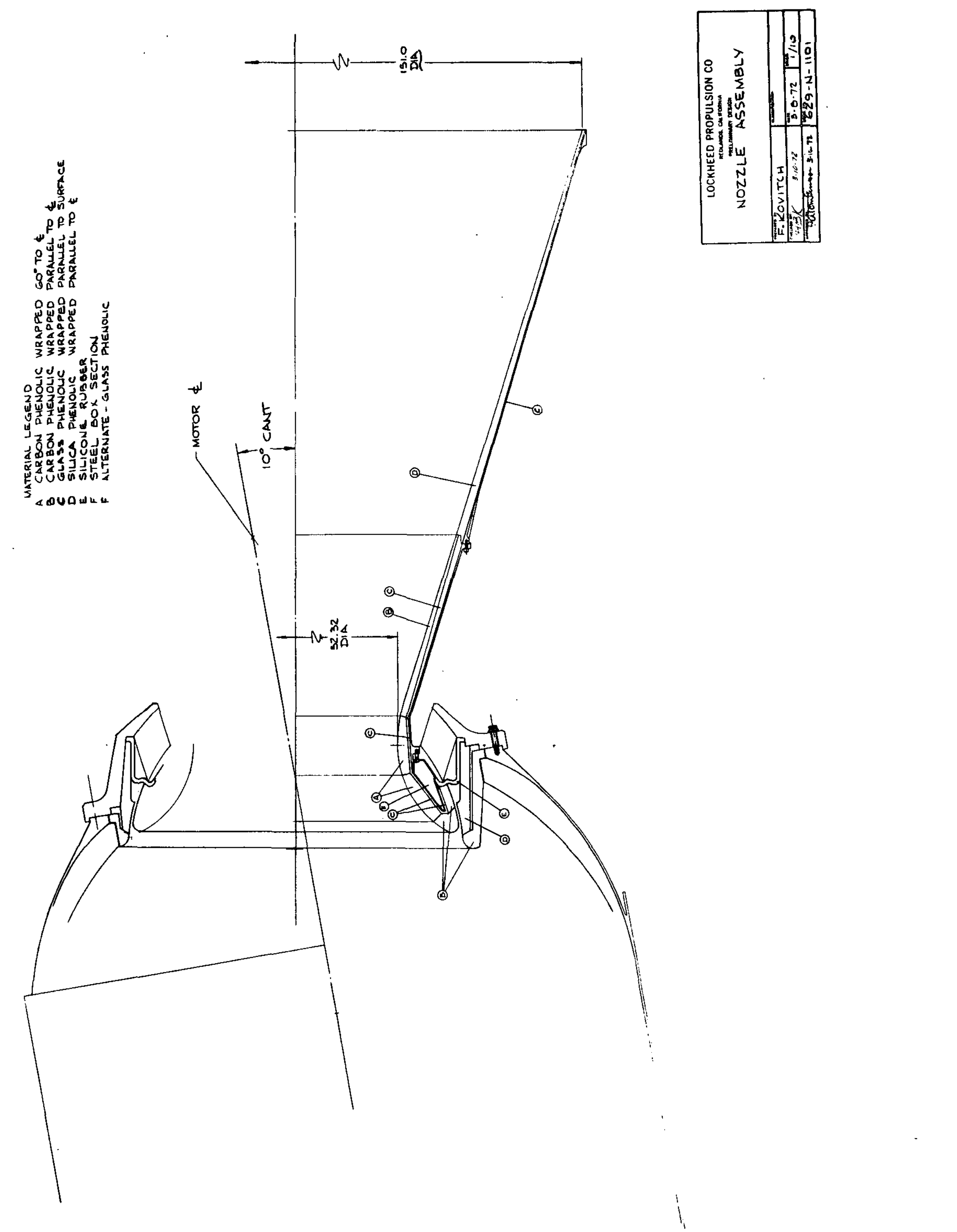
LOW RISK, PROVEN MATERIALS	LOW COST MATERIALS
156 AND 260-INCH SRM, TITAN IIID	DEVELOPMENT TESTS ONLY
CARBON PHENOLIC	CANVAS PHENOLIC
SILICA PHENOLIC	LOW PURITY CARBON
GLASS PHENOLIC	ADDITIONAL DEVELOPMENT SRM's REQUIRED

COST

UNIT COSTS	303K	237K
PROGRAM COSTS	267.5M	209.3M
PROGRAM SAVINGS		58.2M

NOZZLE APPROACH SELECTED




MATERIAL LEGEND
A CARBON PHENOLIC WRAPPED 60° TO ℄
B CARBON PHENOLIC WRAPPED PARALLEL TO ℄
C GLASS PHENOLIC WRAPPED PARALLEL TO SURFACE
D SILICA PHENOLIC WRAPPED PARALLEL TO ℄
E SILICONE RUBBER
F STEEL BOX SECTION
F ALTERNATE - GLASS PHENOLIC
MOTOR ℄
10° CANT
151.0 DIA
52.32 DIA
LOCKHEED PROPULSION CO
NOZZLE ASSEMBLY
F. KOVITCH
629-N-1101

3.3 IGNITER

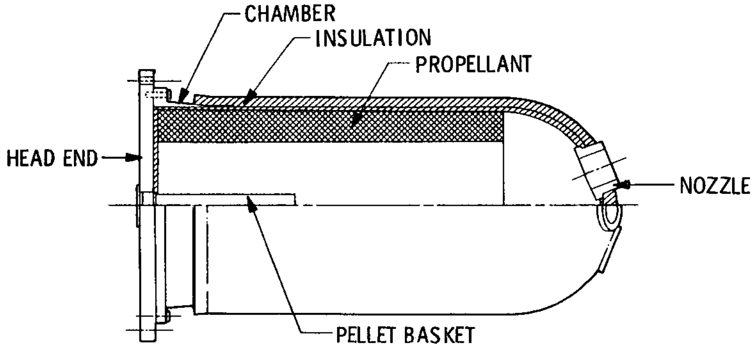
The chart below indicates some of the tradeoff considerations entering into the selection of the baseline igniter design. The chart following presents the baseline design together with the selection rationale for each of the major features incorporated into the design. It will be noted that the basis for selection of each subcomponent (as it was for the SRM in its entirety) was demonstrated design approach and experience to assure minimum-risk development.

IGNITION ALTERNATIVES CONSIDERED

- o HEAD END PYROGEN *
- o AFT END PYROGEN
  - o SCALE-UP LPC 156 INCH IGNITER *
  - o SCALE-UP LPC APOLLO IGNITER
    - o LPC-580A P-BAN PROPELLANT (22 POINT STAR) *
    - o LPC-638A P-BAN PROPELLANT (8 POINT STAR)
    - o LPC-684A HTPB PROPELLANT (8 POINT STAR)
      - o D6AC STEEL CHAMBER WITH WELDED HEMISPHERICAL END *
      - o D6AC STEEL CHAMBER WITH BOLTED-ON 4340 END PLATE
      - o 4130 STEEL CHAMBER WITH WELDED HEMISPHERICAL END

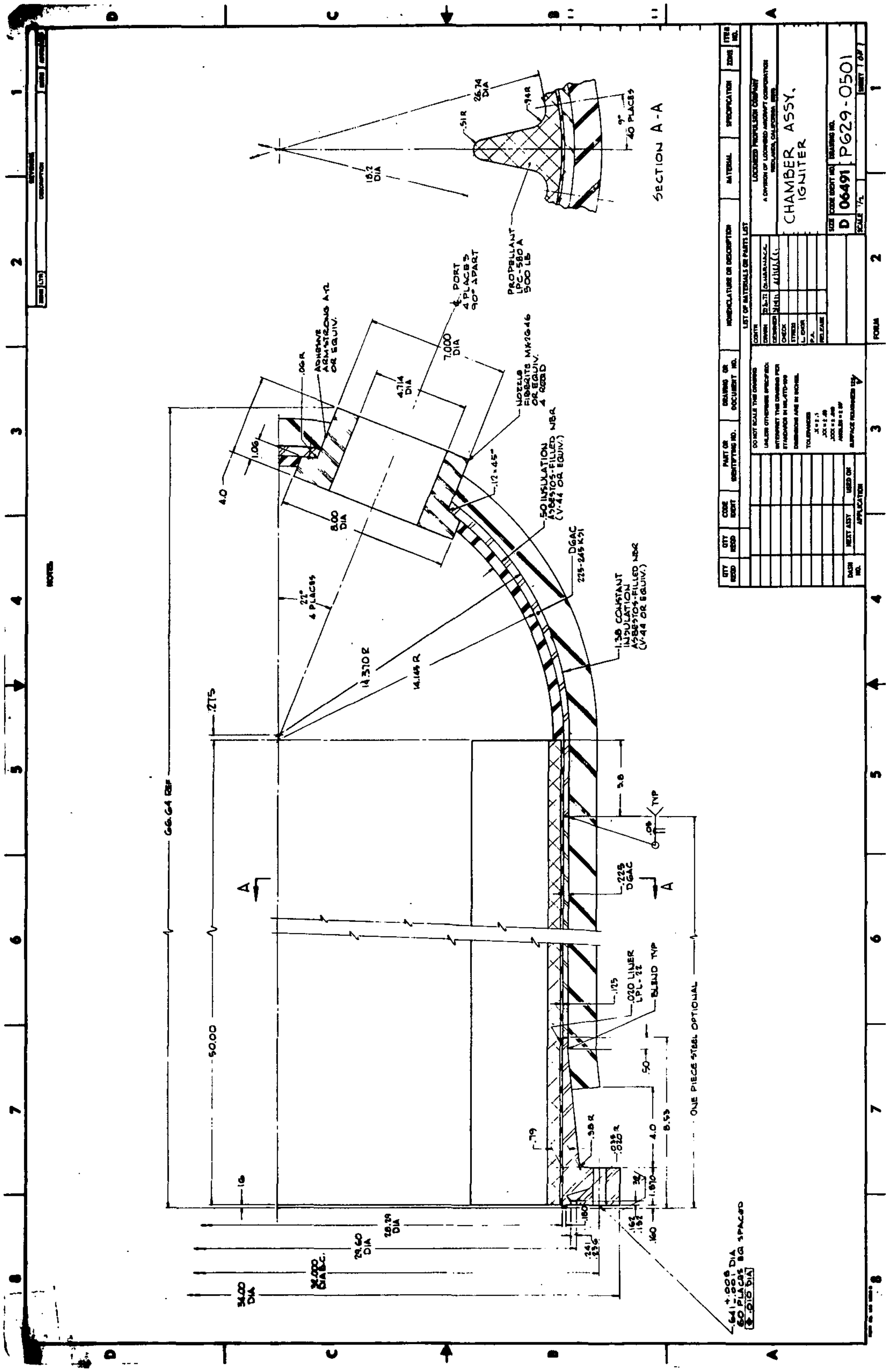
* SELECTED FOR BASELINE DESIGN

156-7 IGNITER DESIGN



FEATURE	SELECTION RATIONALE
HEAD END PYROGEN	TITAN III, 156-IN. SRM
D6AC CASE	TITAN III, MINUTEMEN, SRM
NBR INSULATION	TITAN III, 156-IN. SRM
PBAN PROPELLANT	TITAN III, 156-IN. SRM
DUAL EBW INITIATORS	POLARIS, POSEIDON, SPRINT REDUNDANCY, SAFETY

20


CHAMBER ASSY.
IGNITER
D 06491 P629-0501
SECTION A-A
PROPELLANT LPC-580A 500 LB
ONE PIECE STEEL OPTIONAL
LOCKHEED PROPULSION COMPANY

3.4 INTERNAL INSULATION

Insulation selection involved consideration of both materials and fabrication methods. Presented in this subsection are details of the approaches considered and of the design chosen. Maximum emphasis was placed on the selection of proven, low-cost, state-of-the-art materials and fabrication processes that could be supported by the test of successful production experience and reliable flight performance.

INTERNAL INSULATION - APPROACHES CONSIDERED

FABRICATION METHOD	MATERIALS
LAY-UP IN PLACE/PRESSURE CURE	NITRILE BUTADIENE RUBBER (NBR) WITH ABLATIVE RESISTANT FILLERS - FLIGHT QUALIFIED
CAST-IN-PLACE TROWEL-IN-PLACE SPRAY-IN-PLACE	VARIOUS MASTIC RUBBERS WITH ABLATIVE RESISTANT FILLERS PROBLEMS ENCOUNTERED WITH PAST LARGE SOLID MOTOR APPLICATIONS

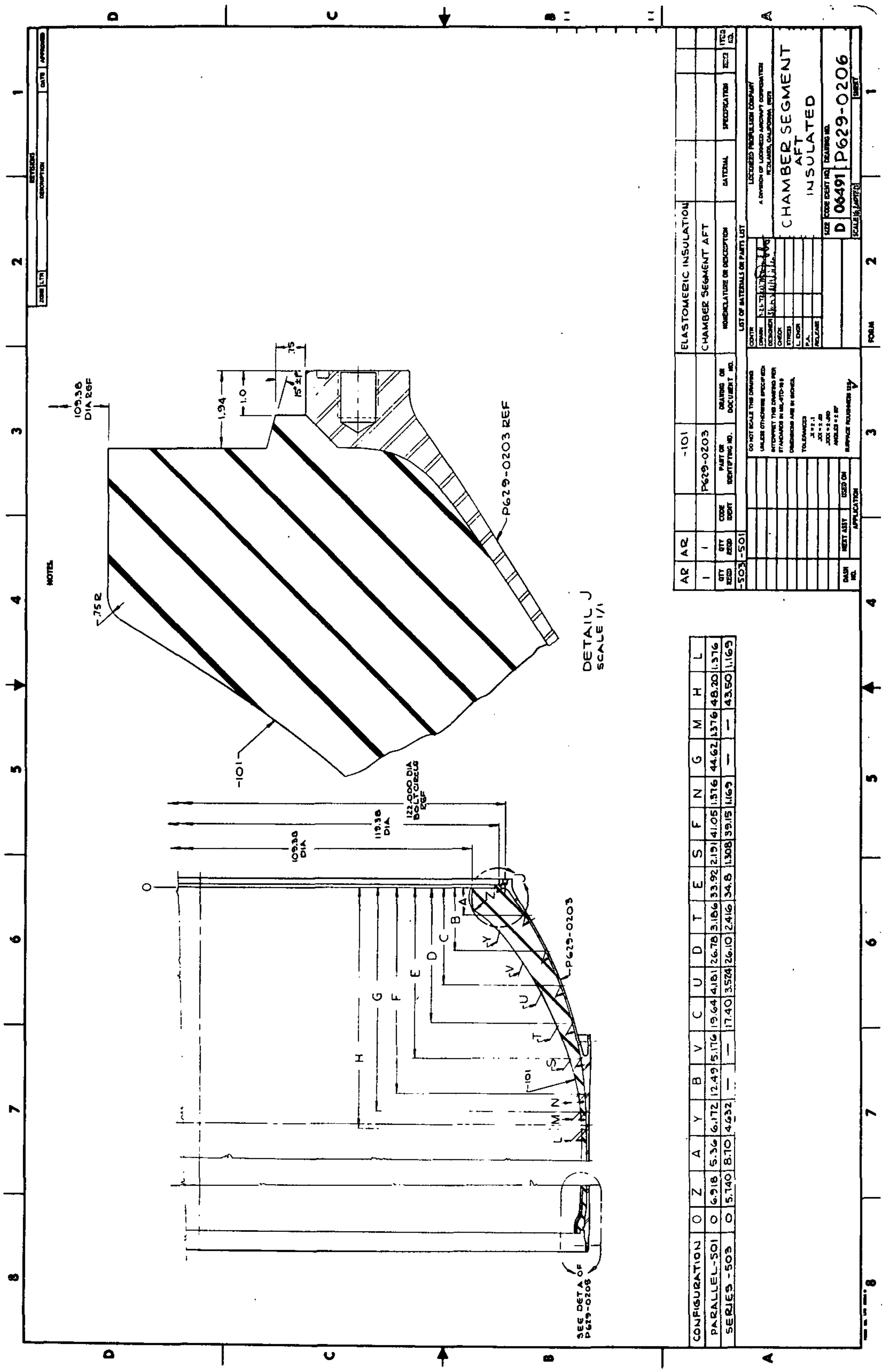
INTERNAL INSULATION SELECTED

DESIGN

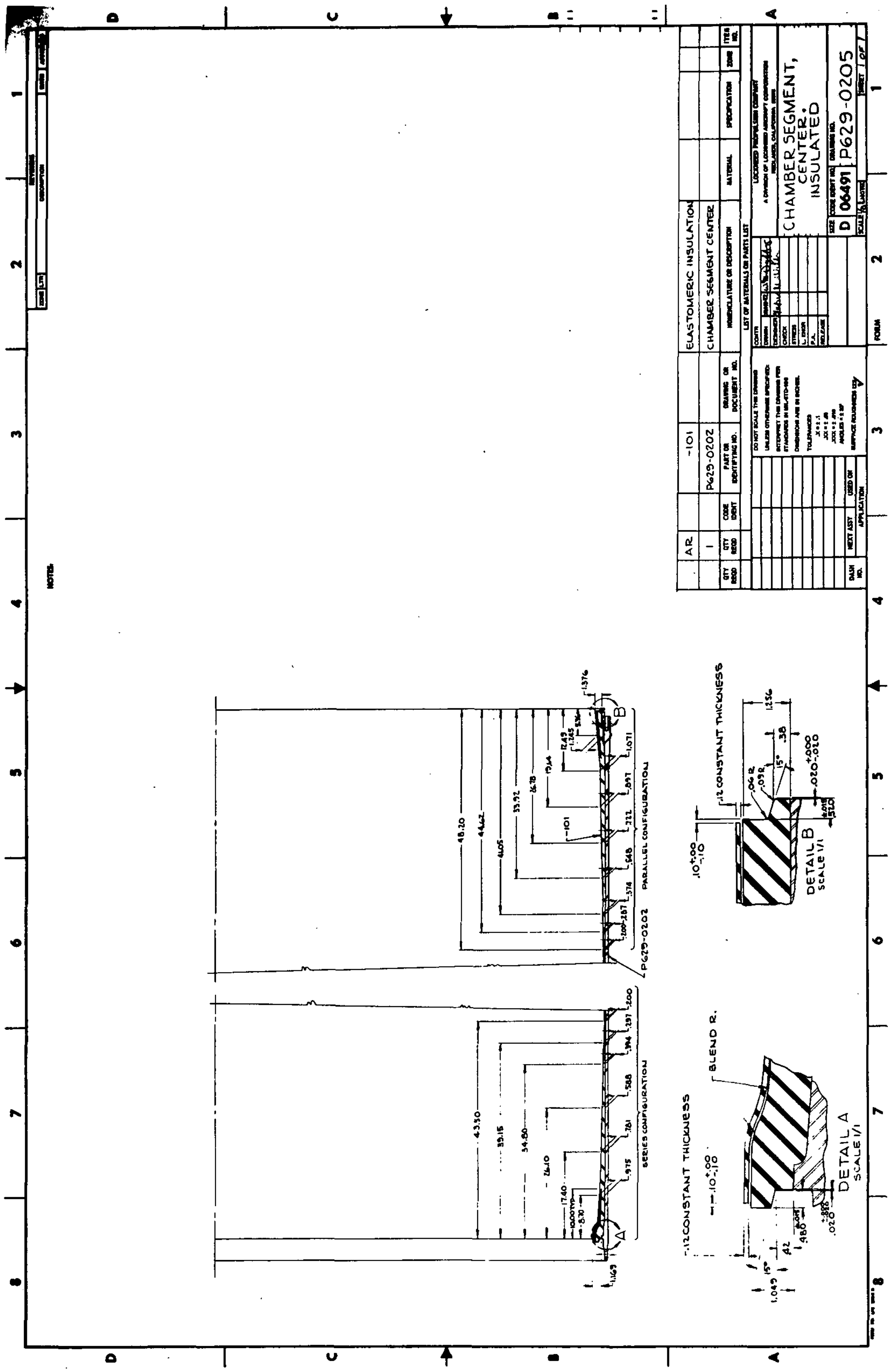
- SILICA-FILLED NBR/ASBESTOS-FILLED NBR
- CALENDERED SHEET STOCK
- AUTOCLAVE CURE IN PLACE

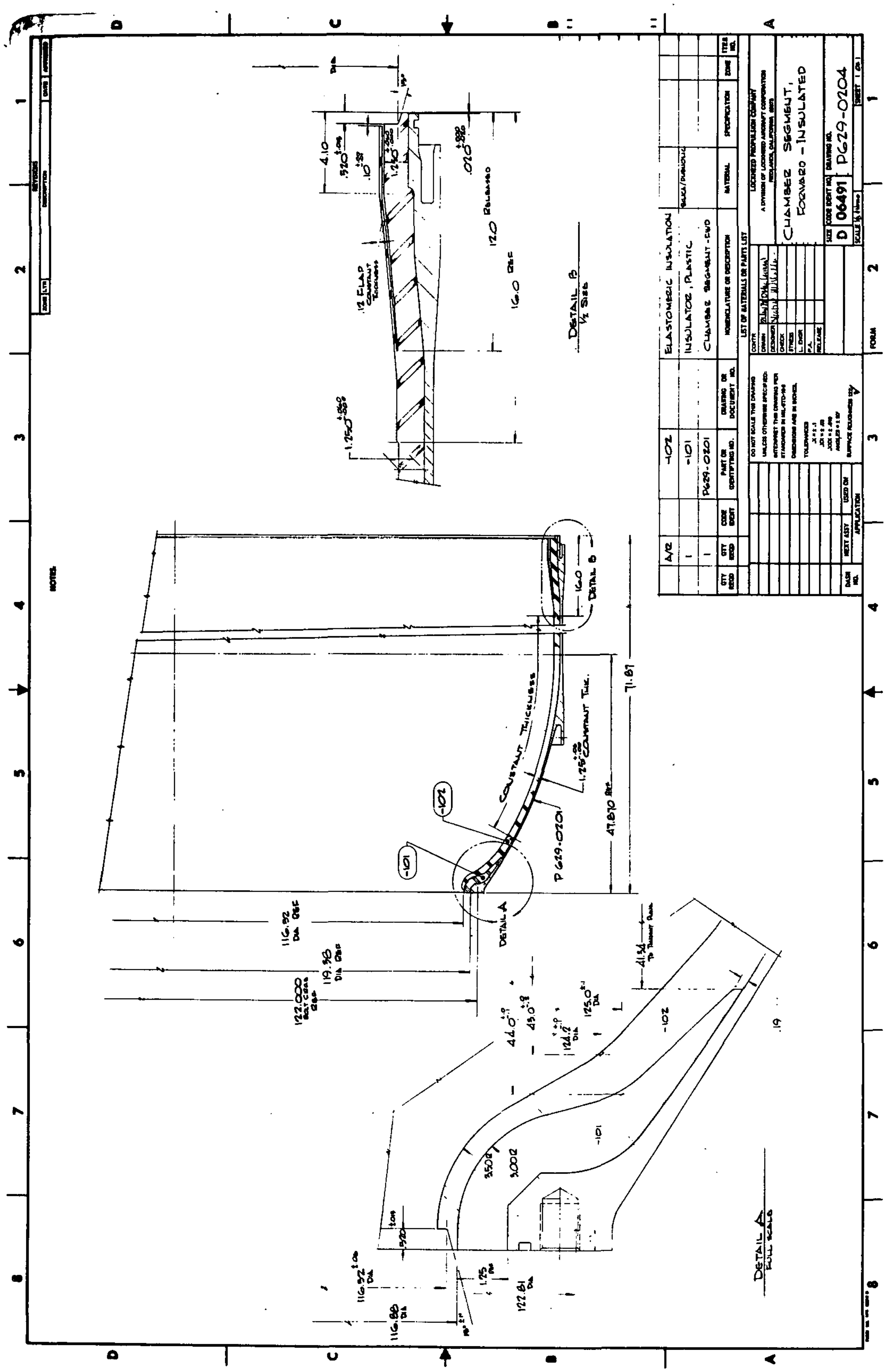
PRODUCTION EXPERIENCE

- PROVEN COMPATIBLE WITH PBAN PROPELLANT
- PROVEN RELIABILITY IN PRODUCTION SRM PROGRAMS
  - LPC 156-INCH SRM'S
  - 260-INCH SRM
  - MINUTEMAN
  - TITAN III-D
  - POLARIS/POSEIDON



CONFIGURATION	O	Z	A	Y	B	V	C	U	D	T	E	S	F	N	G	M	H	L
PARALLEL-501	O	6.918	5.36	6.172	12.49	5.176	19.64	4.181	26.78	3.186	33.92	2.191	41.05	1.376	44.62	1.376	48.20	1.376
SERIES-503	O	5.140	8.10	4.632	—	—	17.40	3.574	26.10	2.416	34.8	1.308	39.15	1.169	—	—	43.50	1.169


CHAMBER SEGMENT, CENTER, INSULATED
D 06491 P629-0205
ELASTOMERIC INSULATION
CHAMBER SEGMENT CENTER
-101
P629-0202
AR
1
LIST OF MATERIALS OR PARTS LIST
SERIES CONFIGURATION
PARALLEL CONFIGURATION
DETAIL A
SCALE 1/1
DETAIL B
SCALE 1/1
.12 CONSTANT THICKNESS
BLEND R.
NOTES:


ELASTOMERIC INSULATION
INSULATOR, PLASTIC
CHAMBER SEGMENT-FWD
-102
-101
P629-0201
LIST OF MATERIALS OR PARTS LIST
LOCKHEED PROPULSION COMPANY
CHAMBER SEGMENT, FORWARD - INSULATED
D 06491 P629-0204
DETAIL A FULL SCALE
DETAIL B 1/2 SIZE
CONSTANT THICKNESS
P629-0201
71.87
47.870 REF

3.5 PROPELLANT

SRM PROPELLANT TRADE-OFFS

REQUIREMENTS

SELECT PROPELLANT FORMULATION TO MEET BURN RATES OF 0.40 TO 0.47 AT 1000 PSI SUITABLE FOR SRM DESIGNS. ESTABLISH MATERIAL AND PRODUCTION COSTS.

APPROACHES CONSIDERED

MODIFIED LPC-580 PBAN PROPELLANT (87% SOLIDS/18% Al) WITH BURN RATE REDUCED BY REDUCTION IN BURN RATE CATALYST AND CHANGE IN OXIDIZER PARTICLE SIZE DISTRIBUTION.

LOW BURN RATE THIOKOL 86% SOLIDS 156-INCH MOTOR PBAN PROPELLANTS TPH 8163 AND TPH-1011.

88% SOLIDS R-45 M HTPB PROPELLANT

90% SOLIDS R-45 M HTPB PROPELLANT

PROPELLANT TRADE-OFFS

DESIGN	LPC-580 PBAN	LPC-629-90 HTPB
PERCENT SOLIDS	87	90
PERCENT ALUMINUM	18	20
PROPELLANT WT/LAUNCH	2.46×10^6	2.37×10^6
PRODUCTION EXPERIENCE	WELL CHARACTERIZED FOR SRM's - 156-INCH SRM - 260-INCH SRM RELATED PRODUCTION EXPERIENCE - TITAN III-D - MINUTEMAN STAGE I	DEVELOPMENT ONLY
COST		
$ PER POUND OF MATERIAL	0.263	0.255
$ PER LAUNCH	0.647 M	0.603 M
$ SAVINGS FOR PROGRAM (EXCLUDING DDT & E)		19.6 M

PROPELLANT SELECTED

SELECTED APPROACH

1. MODIFIED LPC-580 PBAN PROPELLANT (87% SOLIDS/18% Al/0.5% Fe_2O_3) ESTABLISHED AS BASELINE.

   PBAN PROPELLANTS REPRESENT PROVEN 156-INCH MOTOR TECHNOLOGY

   87% SOLID PROPELLANTS (LPC-580 AND THIOKOL TPH-1115) DEMONSTRATED IN 156-INCH MOTORS HAVE LOWEST MATERIAL COSTS IN PBAN PROPELLANT SERIES.

   MAXIMUM COST CREDIBILITY DUE TO HISTORY AND EXPERIENCE AT LPC.

2. FURTHER DEVELOPMENT OF 88 - 90% R-45 HTPB PROPELLANTS RECOMMENDED.

   MODEST PERFORMANCE GAINS

   MINOR COST REDUCTION

   IMPROVED PROCESSING CHARACTERISTICS

   TECHNOLOGY APPEARS ENTIRELY FEASIBLE (cf. CONTRACT NAS 3-12061)

   HIGH LEVEL OF DOD SUPPORT FOR HTPB RESEARCH AND TECHNOLOGY

PROPELLANT SELECTION

The baseline propellant selected for this study is a minor modification of the 87 wt% solids LPC-580 PBAN propellant developed specifically for, and used in, 156-inch large solid motors manufactured and tested at Lockheed Propulsion Company. The LPC-580 propellant system was developed with the objectives of high performance, low material cost, and reliable and low cost processing under conditions necessary for the manufacture of large solid motors. It has been thoroughly characterized with respect to physical properties, ballistic properties, optimum cure cycles, burn rate control parameters, bonding and aging behavior, etc, as well as fully demonstrated in the successful processing and test-firing of 156-5 and 156-6 large solid motors. It therefore provides an established and credible baseline for the design and costing of the SRM Shuttle Booster.

The LPC-580A propellant was tailored to a burn rate of 0.86 in./sec at 1000 psi in accordance with the ballistic requirements of the 156-inch motor demonstration program. Since the SRM designs herein described require burning rates in the range of 0.40 to 0.47 in./sec at 1000 psi, minor modification of the formulation is required to provide the lower burning rate. This tailoring will be accomplished by (1) reducing the burn rate catalyst level from 1.5 to 0.5 wt% and (2) reducing the ground oxidizer content of the formulation from 32 to approximately 20 wt%. The reduction in burn rate catalyst results in a moderate increase in propellant specific impulse, as indicated in the following table, which compares the formulation and properties of LPC-580A and LPC-580M propellants.

Although this modified version of LPC-580 propellant has not been characterized, it may be noted that the reduction in the ratio of ground to unground oxidizer is in the direction of somewhat improved solids packing. Therefore, in accord with solid propellant formulation principles and experience, this change will provide improvement in both propellant mechanical properties and processing viscosities, as well as a minor reduction in processing cost. Thus, there is absolutely no risk associated with the modification required to the basic LPC-580 propellant system for SRM use.

LPC-580 PROPELLANT PROPERTIES

LPC-580A

Ingredient	Content (wt%)
PBAN Binder	13.10
Ballistic Modifier	1.50
Aluminum	18.00
NH_4ClO_4, 180μ	34.37
NH_4ClO_4, 6μ	33.03

Theoretical Values	Value
I^0_{sp} (sec)	261.2
Density (lb/in.^{3})	0.0649
T_c (^{0}F)	5873

	Value
Delivered $I_{sp}{}^0_{1000}$ (sec) (156-5)	250.6
r_b, 1000 psi (in./sec)	0.86
n	0.30
π_k (%/^{0}F)	0.15

Mechanical Properties

Test Temperature (^{0}F)	Maximum Stress (psi)	Strain at Maximum Stress (%)
-40	800	10
0	355	18
70	146	23
140	76	22

LPC-580M

Ingredient	Content (wt%)
PBAN Binder	13.00
Ballistic Modifier	0.50
Aluminum	18.00
NH_4ClO_4, 180μ	48.50
NH_4ClO_4, 6μ	20.00

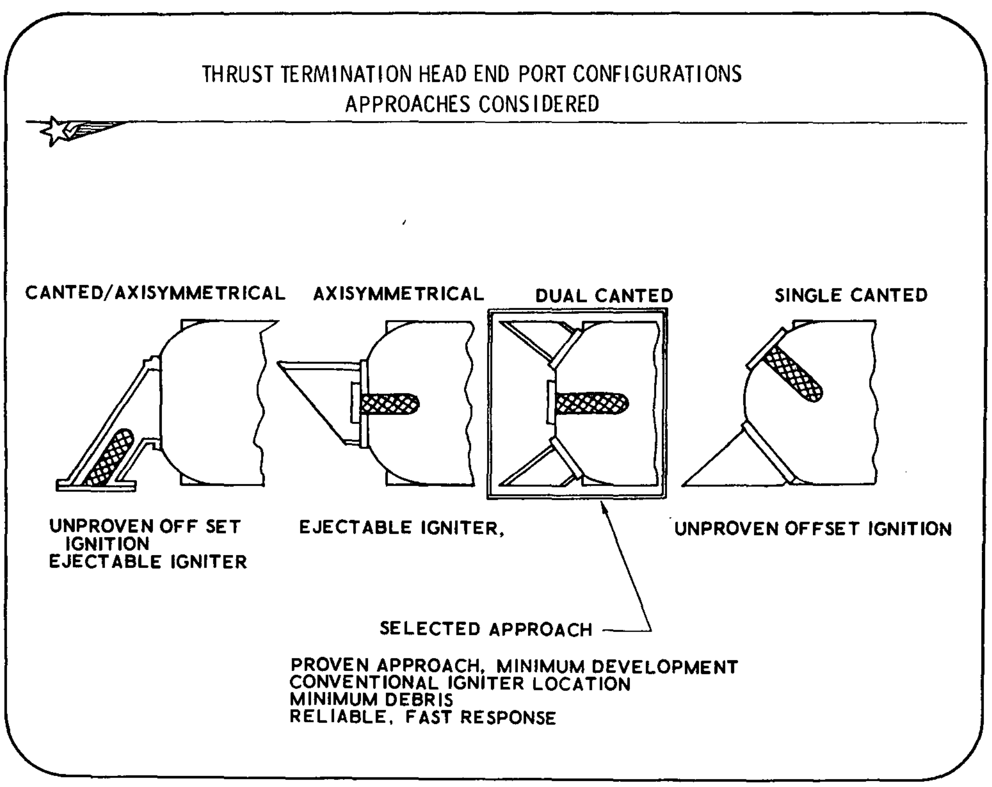
Theoretical Values	Value
I^0_{sp} (sec)	262.6
Density (lb/in.^{3})	0.0646
T_c (^{0}F)	5907

	Value
Delivered $I_{sp}{}^0_{1000}$ (sec) (SRM)	252.0 (predicted)
r_b, 1000 psi (in./sec)	0.40
n	0.30
π_k (%/^{0}F)	0.15

3.6 THRUST TERMINATION SYSTEM

This subsection describes the approaches considered in the selection of a thrust termination system, the geometric details of the candidate systems, and the advantages of the selected system. Function time (that is, the time required to fully open the termination port after command signal) and the maximum tolerance between the time of opening of the two ports were estimated from actual test data of flight motors shown in the table below:

Motor	Function Time (msec)	Maximum Operating Time Tolerance Between Two Ports (msec)
Polaris Stage II	3.0	8.0
Poseidon Stage II	1.5	3.0
Minuteman Stage III	0.5	0.25
120-Inch SRM	1.6	4.0
156-7 SRM (estimated)	2.0	4.0



THRUST TERMINATION

OBJECTIVES

PROVIDE RELIABLE NEUTRALIZATION AT ANY BURN TIME
ELIMINATE DEBRIS IMPACT POSSIBILITY
MINIMIZE

- PRESSURE ENVIRONMENT ON ADJACENT BODIES
- THERMAL INPUT TO ADJACENT BODIES
- TIME TO NEUTRALIZATION

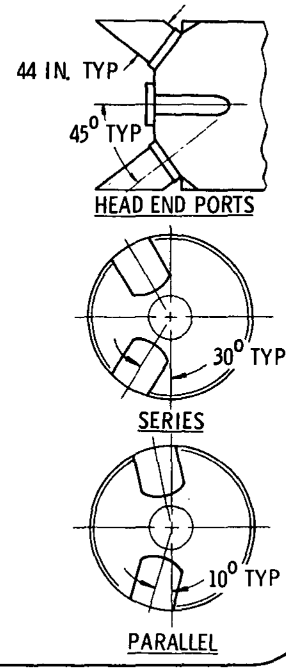
CONSTRAINTS

PROXIMITY OF SRMs TO ADJACENT BODIES

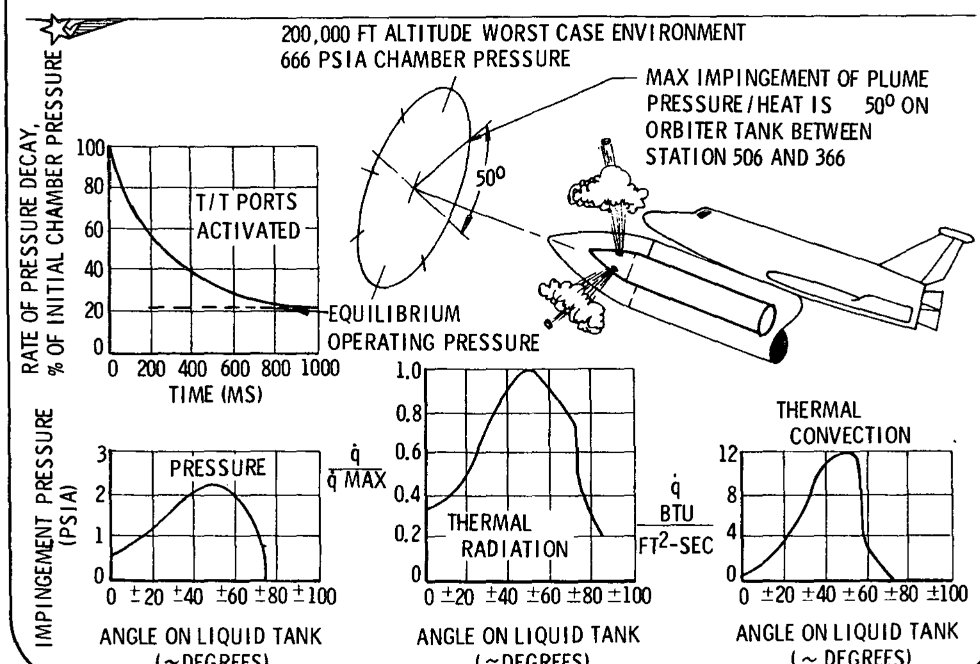
APPROACH

HEAD END PORTS

DEMONSTRATED TECHNOLOGY



PARALLEL BURN SRM THRUST TERMINATION



3.7 THRUST VECTOR CONTROL SYSTEM

This subsection contains a description of the tradeoffs and selection criteria that led to the choice of the Lockseal flexible joint thrust vector control system, a design that has been used with outstanding success in 100 Poseidon test flights. A schematic of the thrust vector control actuation system showing the required tankage, valving, and servo mechanisms is included, followed by a complete summary of system characteristics and performance. The key to the selection of the Lockseal concept was potential low cost and thoroughly reliable performance that could be substantiated by wide industry experience.

BASELINE TVC SYSTEM COMPARISONS

MECHANICAL JET SPOILER SYSTEM

PRO	CON
SIMPLE	LOW PERFORMANCE (<10°)
INEXPENSIVE	LARGE THRUST LOSSES
	SEVERE MAT'L DEVELOPMENT PROBLEMS
	REQUIRES MOTOR TESTS TO DEVELOP SYSTEM

MOVABLE NOZZLE SYSTEMS

PRO	CON
SIMPLE	LARGE INERTIA LOADS TO MOVE
INEXPENSIVE	
CAN DEVELOP SYSTEM WITHOUT MOTOR TESTS	
HIGH PERFORMANCE (>20°)	
SIZE INDEPENDENT OF SIDE IMPULSE	
EXCELLENT GROWTH POTENTIAL	
NO THRUST LOSSES	
MIN. DEVEL TIME AND RISK	

LIQUID INJECTION SYSTEM

PRO	CON
THRUST AUGMENTATION FROM INJECTANT	COMPLEX (# OF COMPONENTS)
	LOW PERFORMANCE (<6°)
	VERY INEFF. AT ANGLES >4°
	SIZE DEPENDENT ON SIDE IMPULSE
	EXPENSIVE
	LITTLE GROWTH POTENTIAL
	REQUIRES MOTOR TESTS TO DEVELOP

GAS INJECTION SYSTEM

PRO	CON
THRUST AUGMENTATION FROM INJECTANT	COMPLEX (# OF COMPONENTS)
BETTER PERFORMANCE THAN LITVC	MATERIAL PROBLEMS FOR HOT GAS SYSTEM
RELATIVELY LOW WEIGHT	LOW PERFORMANCE (<8°)
	REQUIRES MOTOR TESTS TO DEVELOP
	SIZE DEPENDENT ON SIDE IMPULSE EXCEPT FOR CHAMBER BLEED SYSTEM
	VERY INEFFICIENT AT ANGLES >4°

OMNIAXIS MOVABLE NOZZLE SYSTEM COMPARISON

	PRO	CON
GIMBALLED	CONSIDERABLE INDUSTRY EXPERIENCE NO PIVOT POINT MOVEMENT	TORQUE REPRODUCIBILITY RELATIVELY EXPENSIVE RELATIVELY HEAVY FOR LARGE MOTORS
BALL AND SOCKET	LOW COST POTENTIAL NO PIVOT POINT MOVEMENT	RELATIVELY HIGH TORQUE TORQUE REPRODUCIBILITY LITTLE INDUSTRY EXPERIENCE
FLUID BAG	LOW ACTUATION TORQUE LOW COST POTENTIAL	LITTLE INDUSTRY EXPERIENCE LITTLE DEMONSTRATED PERFORMANCE SIGNIFICANT PIVOT POINT MOVEMENT
FLEXIBLE SEAL	SEALED WORKING SURFACE - NO SLIDING SEALS (RELIABILITY) LOW COST POTENTIAL CONSIDERABLE INDUSTRY EXPERIENCE REPRODUCIBLE AND PREDICTABLE PERFORMANCE	SMALL PIVOT POINT MOVEMENT

ACTUATION SYSTEM COMPARISON

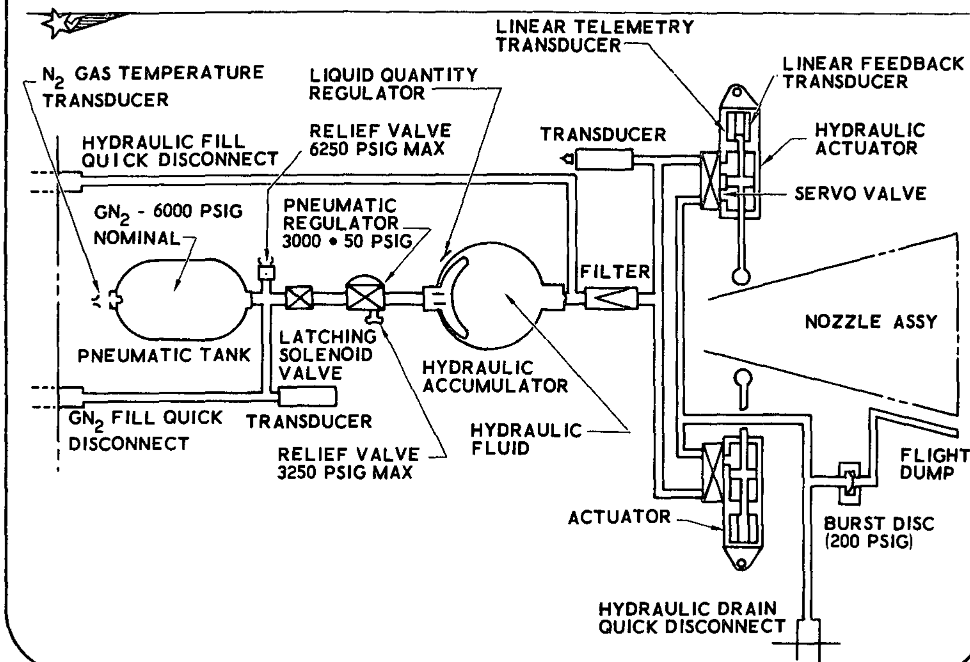
	ADVANTAGES	DISADVANTAGES
HYDRAULIC ACTUATION:	STATE-OF-ART (LOW RISK/ MAXIMUM RELIABILITY) INDUSTRY WIDE EXPERIENCE MAXIMUM REUSEABLE COMPONENTS PREFIRE SYSTEM CHECK FEASIBLE (RELIABILITY)	FLUID LEAKAGE
PNEUMATIC ACTUATION:	POTENTIAL LOW COST PRODUCTION HIGH PERFORMANCE	LITTLE INDUSTRY EXPERINECE PREFIRE SYSTEM CHECK NOT FEASIBLE
SERVO NOZZLE CONTROL ACTUATION:	POTENTIAL LOW-COST PRODUCTION PARTIAL PREFIRE CHECK FEASIBLE (RELIABILITY)	NO EXPERIENCE REQUIRES MOTOR TESTING TO DEVELOP SYSTEM

TVC SYSTEMS CONSIDERED AND SELECTED APPROACH

TVC TYPES	MOVABLE NOZZLE TYPES	ACTUATION SYSTEM	POWER SUPPLY SYSTEM
MOVABLE NOZZLE*	GIMBALLED	HYDRAULIC*	WARM GAS TURBO PUMP
LIQUID INJECTION	FLEXIBLE SEAL*	PNEUMATIC (WARM OR COLD)	WARM GAS BLOW-DOWN
GAS INJECTION	FLUID BAG	SERVO NOZZLE CONTROL	COLD GAS BLOW-DOWN*
MECHANICAL INTERFERENCE	BALL AND SOCKET		

*SELECTED APPROACH

THRUST VECTOR CONTROL SYSTEM SCHEMATIC



TVC SYSTEM PERFORMANCE AND CHARACTERISTICS

PERFORMANCE CHARACTERISTICS:

VECTOR ANGLE	10°
VECTOR RATE	15°/SECOND
DUTY CYCLE	10 10° SINE WAVE CYCLES

WEIGHT SUMMARY:

NOZZLE	17,004 LB
FLEXIBLE SEAL	8500 LB
SERVO ACTUATORS (2)	1060 LB
COLD GAS BLOW-DOWN° POWER SUPPLY	8940 LB
TOTAL	35,504 LB

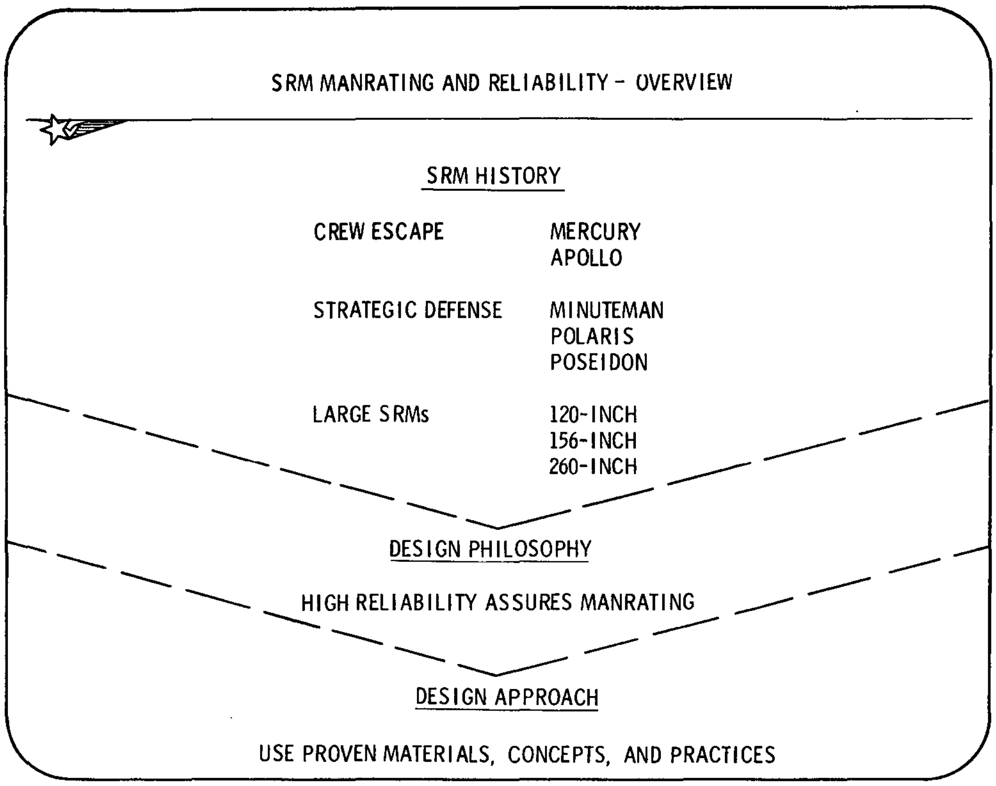
TORQUE CHARACTERISTICS:

FLEXIBLE SEAL AT 10°	2.1×10^6 IN.-LB
THERMAL BOOT AT 10°	0.5×10^6 IN.-LB
VISCOUS AT 15°/SECOND	0.8×10^6 IN.-LB
INTERNAL AERODYNAMIC DUE TO 10° VECTOR ANGLE	1.1×10^6 IN.-LB
INTERNAL AERODYNAMIC DUE TO 10° CANT ANGLE	1.1×10^6 IN.-LB
EXTERNAL AERODYNAMIC (ASSUMED TO BE SHIELDED)	-0-
LATERAL ACCELERATION AT 0.5 G's	0.93×10^6 IN.-LB
AXIAL ACCELERATION NO CANT AT 3 G's AND 10° VECTOR ANGLE	0.96×10^6 IN.-LB
AXIAL ACCELERATION WITH 10° CANT AND 3 G's	0.96×10^6 IN.-LB
TOTAL WITHOUT CANT NOZZLE	6.4×10^6 IN.-LB
TOTAL WITH 10° CANT NOZZLE	8.45×10^6 IN.-LB

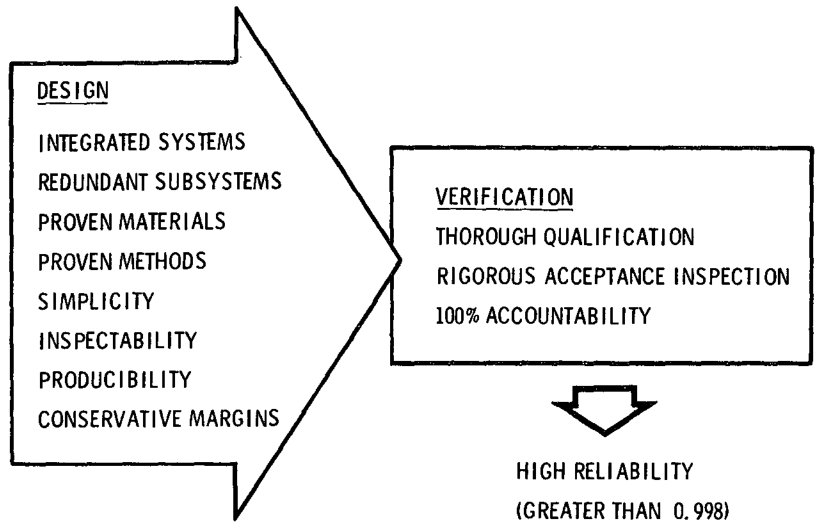
INCLUDES 1.5 FACTOR OF SAFETY ON CAPACITY AND 2.0 SAFETY FACTOR ON PRESSURE TANKS

3.8 MAN-RATING AND RELIABILITY

Lockheed Propulsion Company's design philosophy and approach to the design and development of a man-rated SRM are presented in the following pages. Following this overview, a more detailed analysis lists the SRM components in terms of available and defined approaches to the achievement of a man-rated system. An exposition of required design practices concludes the discussion.



SRM MANRATING AND RELIABILITY – PROGRAM APPROACH



SRM
MANRATING AND RELIABILITY

DESIGN ELEMENT	APPROACHES AVAILABLE	APPROACHES DEFINED
SRM CASE (PRESSURE VESSEL)	CONSERVATIVE ANALYSIS HOMOGENEOUS STRUCTURE STRUCTURAL EFFICIENCY	ADEQUATE FACTOR OF SAFETY (1.4) PROOFING OF STRUCT. INTEGRITY NO-WELD FABRICATION PROVEN, EFFICIENT JOINT PROVEN, FABRICATABLE MATERIAL
PROPELLANT-BALLISTICS/ INSULATION	STATE OF THE ART PROPELLANT RAW MATERIAL CONTROL IN-PROCESS CONTROL POST-PROCESSING INSPECTION	RIGOROUS SAMPLING PROCEDURES RIGOROUS NDT PROCEDURES PROVEN PROPELLANT/PROCESS
IGNITION	CONSERVATIVE BALLISTICS LOADING PROVEN DESIGNS PROVEN COMPONENTS REDUNDANCY	PROPELLANT LOAD SAFETY FACTOR IS HIGH IGNITER IS A ROCKET MOTOR COMPARABLE TO THOUSANDS PRODUCED IN INDUSTRY TWO EBW'S AND FIRING UNITS
NOZZLE	CONSERVATIVE ANALYSIS HOMOGENEOUS STRUCTURE STATE OF THE ART ABLATIVES STATE OF THE ART FABRICATION PRACTICES	ADEQUATE FACTOR OF SAFETY NO WELD FABRICATION PROVEN MATERIALS USING CURRENT LSM FABRICATION PRACTICES
TVC	STATE OF THE ART FLEXIBLE JOINT	LOCKSEAL IN USE ON MISSILE SATURN QUALIFIED ACTUATORS REDUNDANT VALVES, FUNCTIONAL PARTS, ADEQUATE SAFETY FACTOR (2 TO 2.5)
MISCELLANEOUS SUBSYSTEMS	PROVEN COMPONENTS/METHODS RESERVE COMPLEXITIES FOR ORBITER VEHICLE SIMPLIFIED INTERFACINGS	

SRM MANRATING AND RELIABILITY - DESIGN PRACTICES

SYSTEM ORIENTED SRM DESIGN FEATURES

- THRUST TERMINATION FOR ABORT
- PERFORMANCE ANOMALY SENSORS ON SRM
- SRM "INSTANT TURN-ON" HOLDS LAUNCH COMMIT TO T-1 SECOND
- SRM CASE PROVIDES BASIC BOOSTER STRUCTURE

SRM BASELINE DESIGN APPROACH

- PROVEN CASE DESIGN/MATERIALS/FABRICATION METHOD
- PROVEN PROPELLANT AND INSULATION
- PROVEN NOZZLE DESIGN/MATERIALS
- PROVEN TVC SYSTEM APPROACH
- PROVEN IGNITION SYSTEM
- AVIONICS AND DATA COMPLEXITIES ON ORBITER

SRM SAFETY FACTORS

- 1.4 MEOP ULTIMATE CASE STRENGTH
- 1.15 MEOP PROOF TEST ON CASE
- 2.0 ON NOZZLE ABLATIVES
- 2.0 ON CASE INSULATION
- 2.0 ON TVC PRESSURE TANKS, VALVES
- 2.5 ON TVC PLUMBING

REDUNDANT SUBSYSTEMS

- IGNITION CONTROL AND COMMAND
- THRUST TERMINATION CONTROL AND COMMAND
- TVC ACTUATION/VALVING
- ELECTRICAL CONTROL/DATA SENSING

DEVELOPMENT/QUALIFICATION PROGRAM

- MAJOR COMPONENTS DEVELOPED/QUALIFIED SEPARATELY PRIOR TO FULL SCALE PROGRAM
- PROGRAM OF 5 DEVELOPMENT AND 4 PFRT MOTORS
- PROGRAM OF 3 FACILITY QUALIFICATION MOTORS

SRM RELIABILITY FACTORS

- LARGE SIZE REDUCES VARIABILITY OF PROCESS
- SEGMENTING ALLOWS REJECTING AT MINIMUM COST IMPACT
- SRM SIMPLIFIES HUMAN ELEMENT IN BOOSTER ASSEMBLY
- USE OF PREVIOUSLY QUALIFIED COMPONENTS
- USE OF PROVEN, TESTED METHODS FOR NON-QUALIFIED COMPONENTS OR SUBSYSTEMS

3.9 SUBSYSTEM SAFETY/HAZARD ANALYSIS

The following chart presents the results of a comprehensive analysis of potential SRM subsystem manfunction modes. The effects of each malfunction are described, followed by an itemization of reliability design methods and practices that will prevent or circumvent component failure. Malfunction detection methods are then proposed.

Malfunction	Analysis of Mode	Reliability Assurance	Possible Detection Modes
1. Failure of motor to ignite	Solid propellant will ignite readily when subjected to a combination of heat and pressure for a reasonably short time	1. Dual electrical circuits 2. Dual initiators 3. Sustained impulse to the motor igniter 4. Sustain burning of the igniter charge 5. Conservative design	1. Breakwire across nozzle closure 2. Preset pressure switch to register acceptable ignition pressure limit
2. Propellant grain crack	A propellant grain crack exposes added surface area for burning and premature exposure of the chamber wall	1. Internal port is highest stress in the grain and therefore easily inspected 2. Motor is designed for very low grain stress	Pressure sensor to detect large motor overpressure
3. Motor case overpressure	Additional burning surface causes high operating pressure	1. Safety factor allows for very large increase in surface area without failure 2. Surface area increase necessary to overpressure the chamber is easily detected before firing. 3. Grain stresses are reduced when motor chamber pressurizes and grain burns	Pressure sensor to detect large motor overpressure
4. Joint leakage	Chamber joint leaks during firing because of seal failure	1. Basic joint integrity will be proven during hydrotest of segment 2. Low-pressure leak check will be performed to ensure seal integrity before launch 3. Joint insulation will be designed for 100% safety margin and demonstrated during DDT&E 4. Proven design practice will be applied to joint design	Hot gas leak through O-ring seal can be detected by placing breakwire in groove of case joint immediately adjacent (outside) of O-ring groove
5. Inadvertent ignition	Motor ignites because of stray electrical signal or thermal input	1. High-energy initiation system avoids stray energy problem 2. Motor is sealed with a thermally insulating closure in the nozzle 3. Case thickness and liner insulates grain against accidental heat input to motor case	Thrust neutralization could be immediately established by countdown preset
6. Failure of thrust vector control system	Thrust vector control system failure causes unwanted vector or inoperable system	1. Redundant actuation systems are specified 2. System failure will cause nozzle to return to null 3. Conservative cold-gas design is used	Sensor on cold-gas pressure system and feedback position sensor on nozzle could identify TVC failure. Backup pressure system then returns nozzle to null position
7. Nozzle throat structural failure	Nozzle throat is ejected, leading to low motor pressure, offset thrust vector, and nozzle structural failure	1. Conservative design is specified 2. Proven processes and material are used 3. Very large changes in throat are required to affect motor pressure 4. NDT inspection of the nozzle will be performed	Thermocouple monitoring of the nozzle exterior at selected areas could identify gross loss of throat/exit cone components or Lockseal malfunctioning
8. Unbond of insulation or grain	Unbond provides a path for gases, which leads to unplanned heating of the chamber	1. Grain bond strength is very high relative to stresses through use of proven materials and processes 2. NDT inspection of all insulation bonds will be performed 3. Visual inspection of grain bonds will be performed	Thermocouple monitoring of selected areas of motor exterior could identify unacceptable internal insulation functioning
9. Unbond of Lockseal	Lockseal elements unbond from reinforcements, causing leakage or failure of the seal	1. Lockseal element is in compression during firing 2. Lockseal will be subjected to bench test before installation on nozzle to ensure integrity of bonds 3. Lockseal element will be pressure-tested before installation on motor 4. Unbond of Lockseal will not cause failure	See Item 8 above
10. Inadvertent or nonfunction of thrust neutralization ports	Lack of proper function will cause compromise of abort system or premature thrust neutralization	1. Redundant electrical circuitry 2. High-energy initiation system 3. Redundant initiators 4. Conservative design practice	Breakwire system on interior or exterior of each TT port

Section 4

MOTOR/VEHICLE INTEGRATION

4.1 BASELINE DESIGN: 156-INCH, PARALLEL-BURN BOOSTER VEHICLE

This subsection contains a description of the tradeoff considerations leading to the selection of baseline electrical systems for the ignition, TVC, and instrumentation components. Baseline electromechanical, avionics, and related interface systems are then presented in narrative, chart, and schematic form, together with supporting component selection rationale. The flight control system is described next and is followed finally by charts detailing the proposed launch/flight sequence.

4.1.1 Tradeoffs

CANDIDATES/TRADEOFFS
PARALLEL

IGNITION SYSTEM	LOW VOLTAGE	HIGH VOLTAGE SAFE
	*HIGH VOLTAGE	ELECTRICAL "ARM CONTROL" AND "DISARM"
	CDF/TBI	REMOVAL OF INVERTER
		HIGH VOLTAGE IS LESS SUSCEPTIBLE TO EMI/RFI AND TAMPERING OR SHORTS
		LOW VOLTAGE LESS EXPENSIVE, BUT REQUIRES INDIVIDUAL MECHANICAL SAFE/ARMS
		CDF TO BE USED ONLY IF SIMULTANEOUS MULTIPLE OUTPUTS FROM ONE FIRING UNIT REQUIRED
TVC ELECTRICAL SYSTEM	*D.C. ANALOG	DIGITAL COMPATIBLE WITH COMPUTER LOGIC FOR REDUNDANT "VOTING" TECHNIQUES
	DIGITAL	ANALOG IS LESS EXPENSIVE AND HAS SIMPLER INTERFACE WITH ORBITER
INSTRUMENTATION	PULSE CODE MODULATION	PCM HAS HIGHER DATA RESOLUTION
	*PULSE AMPLITUDE MODULATION	PAM IS LESS EXPENSIVE AND PROVIDES ACCEPTABLE DATA.

*BASELINE SELECTION

4.1.2 Electromechanical, Avionics, and Related Interfaces

Electrical system. Independent electrical systems are provided in the forward end of the motor (high voltage) and the aft end (low voltage) to eliminate the need for a raceway. This is particularly helpful with a multiple segment motor. Umbilicals connecting the SRMs to the orbiter will be located at the forward and aft tank attach points.

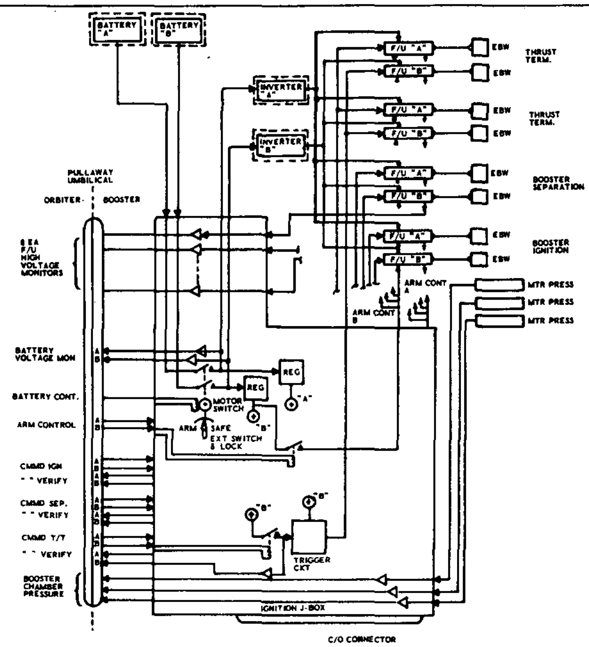
The forward electrical system will provide power for the ignition thrust termination, separation rockets, and motor sensor functions. The aft system will provide command signals for actuation of the TVC system.

Ignition electrical system. The ignition system is entirely dual redundant. The two batteries can be turned on with the motor-driven switch prior to launch. The voltage can then be checked. Next, the arm control is cycled to check the inverters and the firing units. After the firing units are discharged, each orbiter command can be cycled and monitored to verify that it was received by the booster. The firing units are all recharged just prior to launch. All orbiter commands are dual redundant. The motor switch can be manually turned off in case of a hold and locked "safe" while working on the system. Arming control also provides a method of removing the high voltage from the firing units in case of hold. Removing the inverters is the primary method of "safing" during ground handling and checkout.

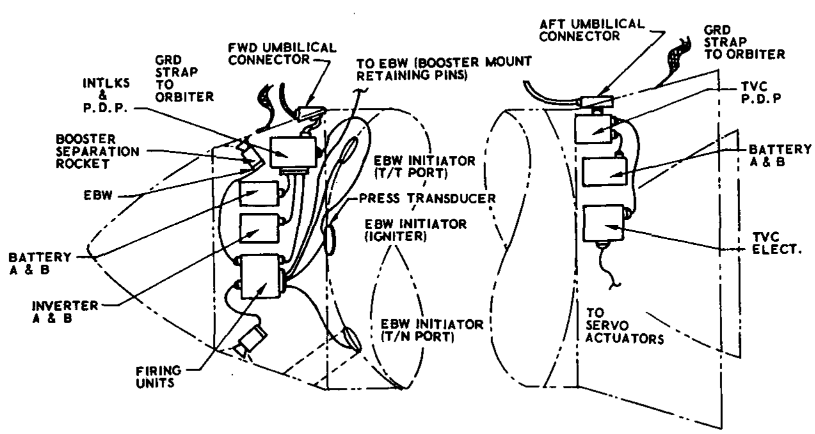
STAGE FORWARD END ELECTRICAL SYSTEM

- REDUNDANT CIRCUITRY AND POWER SUPPLIES THROUGHOUT IGNITION SYSTEM
- ENTIRE SYSTEM QUALIFIED FOR RANGE AND HUMAN SAFETY
- ELECTRICAL ISOLATION FROM ORBITER AND OTHER SRM CIRCUITS
- A3 TYPE INVERTERS AND FIRING UNITS
- ALL MONITOR CIRCUITS PRECONDITIONED, IMPEDANCE ISOLATION, 0 - +5 VOLT SIGNAL CONDITIONED
- 30 DAY WETSTAND BATTERIES (REMOVABLE)
- INVERTERS REMOVED TO PROVIDE "SAFE" FUNCTION
- EXTERNAL BATTERY "ARM/SAFE" SWITCH
- ELECTRICAL "ARM CONTROL" PROVIDES REMOTE DISARM CAPABILITY FOR IGNITION SYSTEMS
- SINGLE POINT GROUND
- EBW HIGH VOLTAGE IGNITION SYSTEM HAS HIGH SAFETY AND RELIABILITY AS PROVEN ON POLARIS/POSEIDON VEHICLES

STAGE FORWARD END ELECTRICAL SYSTEM (PARALLEL BURN)



PARALLEL BURN STAGE ELECTRICAL/ORDNANCE



FEATURES: HIGH VOLTAGE EBW INITIATION SYSTEMS FOR SAFETY
REDUNDANT CIRCUITRY AND POWER SUPPLIES FOR RELIABILITY
INDEPENDENT FORWARD AND AFT SECTIONS
NO RACEWAY

TVC electrical system. The latching solenoid valve and the scavenge pump are operated by ground power. The scavenge pump returns the leakage and used hydraulic fluid during ground test to the reservoir. The solenoid valve is latched to "on" and the system is checked just prior to launch. The Power Control J-Box controls the batteries and is operated by ground power just prior to launch. If required, a flight test instrumentation package will be added to the system to provide a J-Box for telemetry signals. The output will be signal-conditioned and probably multiplexed using a commutator to provide all the signals to the orbiter on one line.

TVC ELECTRICAL SYSTEM
GENERAL CHARACTERISTICS

WET CELL BATTERIES (30 DAY WETSTAND)

REDUNDANCY, MAJORITY VOTING AND FAILURE DETECTION IN TVC SYSTEM

TRIPLE VOTING REDUNDANCY ON SERVO AMPLIFIERS

SINGLE POINT GROUND

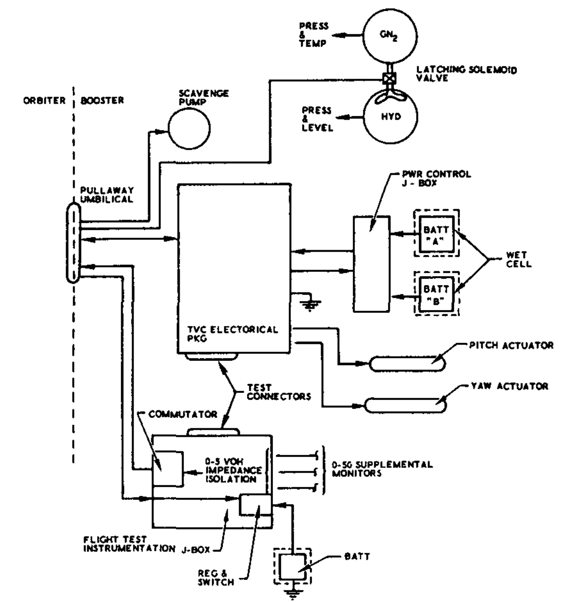
ELECTRICAL ISOLATION FROM ORBITER AND OTHER SRM CIRCUITS

ALL MONITORS CONDITIONED TO 0→5 VOLTS AND IMPEDANCE ISOLATED

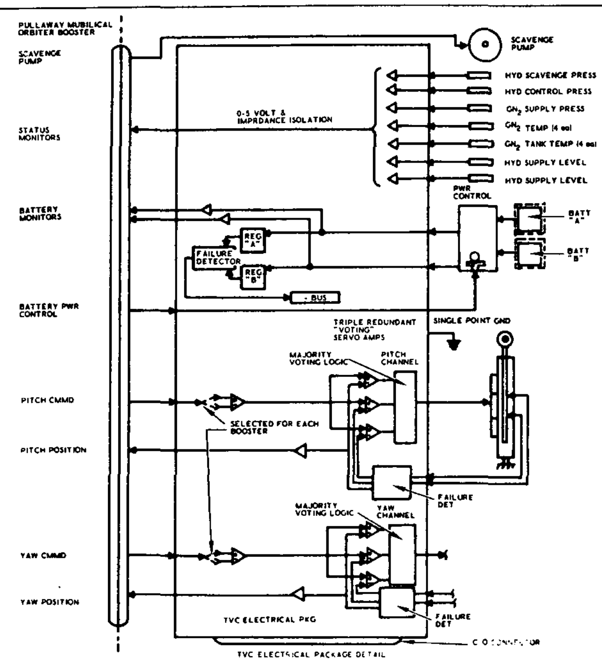
DIRECT BYPASS TO RUN OFF FACILITY HYDRAULIC SOURCE

FLIGHT TEST INSTRUMENTATION J-BOX ON INITIAL FLIGHTS ONLY

TVC ELECTRICAL SYSTEM SCHEMATIC



TVC ELECTRICAL PACKAGE DETAIL



Mechanical design. Mechanical design details of major components of the parallel-burn SRM are presented below and on the following pages.

PARALLEL BURN STAGE MECHANICAL DESIGN
CANDIDATES AND ALTERNATES

ITEM	ALTERNATIVES	SELECTION RATIONALE
NOSE FAIRING/ ADAPTER	CONTOUR - SMOOTH VS DOUBLE CONE	DOUBLE CONE FIXES AERO FLOW SEPARATION POINT
	N/F - SEPARABLE VS FIXED	NO REQUIREMENT IDENTIFIED TO HAVE N/F SEPARATION
TANK ATTACH	THRUST TAKE-OUT - FWD VS AFT	FWD TAKE-OUT SHORTENS ORBITER LOAD PATH
	JETTISON - SLIDING RELEASE VS EXPLOSIVE RELEASE	SLIDING RELEASE SIMPLE, RELIABLE
	BOOSTER PICKUP POINTS - SADDLE (℄) VS ECCENTRIC	SADDLE CONFIGURATION ELIMINATES BOOSTER CASE BENDING MOMENT DUE TO THRUST REACTION
	SEPARATION - JETTISON ROCKETS VS INERTIAL (ABOVE CONCEPTS CONSISTENT WITH TITAN III-C SYSTEM)	ROCKETS PROVIDE MORE RAPID SEPARATION
AFT SKIRT	CONFIGURATION - FLARED VS STRAIGHT	FLARE PROVIDES AERODYNAMIC SHIELD FOR DEFLECTED NOZZLE

STAGE BASELINE DESIGN MECHANICAL FEATURES
PARALLEL BURN

SMALL SRM USED FOR OUTBOARD PITCHING MOMENT AT JETTISON

THRUST TAKE-OUT AT FORWARD ATTACH POINT IS APPLIED AT STRONGEST ORBITER TANK STRUCTURE

MAXIMUM RELIABILITY WITH REDUNDANT EXPLOSIVE SEPARATION BOLT/LINK

INDEPENDENT FORWARD AND AFT ELECTRICAL SYSTEMS AND UMBILICAL CONNECTORS ELIMINATE NEED FOR RACEWAYS

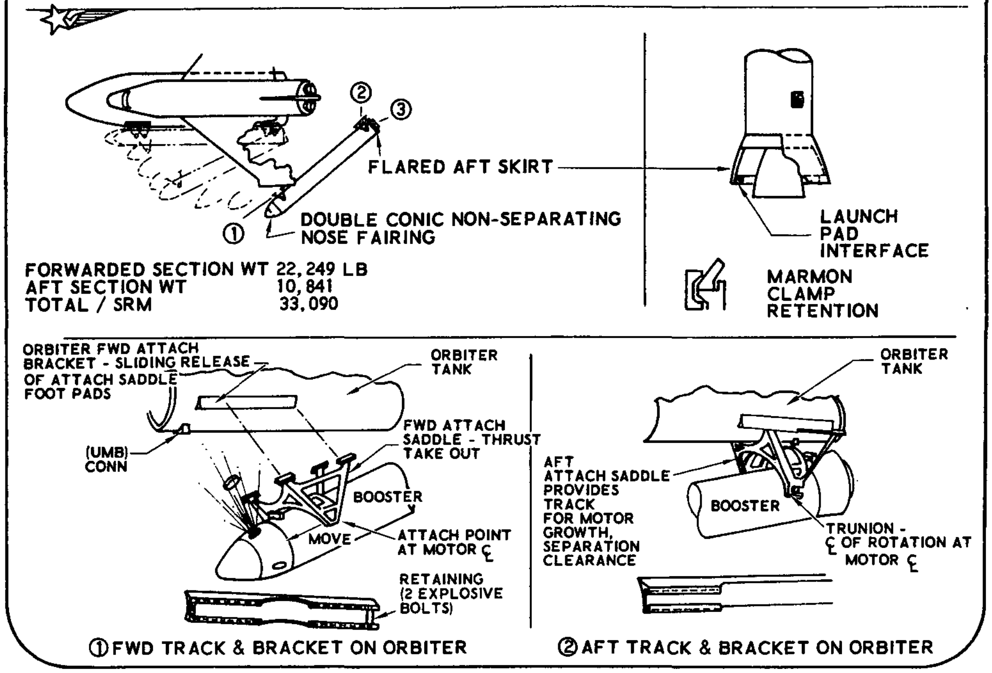
FLARED SKIRT PROVIDES AERODYNAMIC SHIELD FOR NOZZLE, TAKES PAD VEHICLE SUPPORT AND HOLD DOWN LOADS

THRUST TERMINATION STACKS CANTED TO CLEAR ORBITER AND TANK, PROVIDE TUMBLING MOMENT FOR ABORT MODE

SRM CENTERLINE THRUST TAKEOUT MINIMIZES MOTOR CASE BENDING LOADS

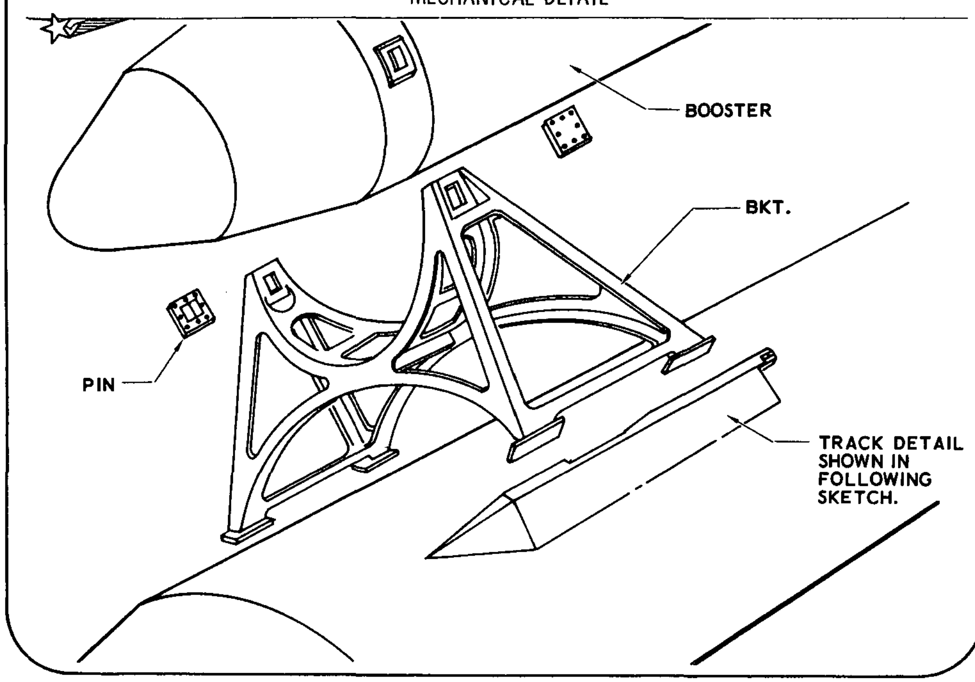
STAGE, PARALLEL BURN

MECHANICAL DESIGN

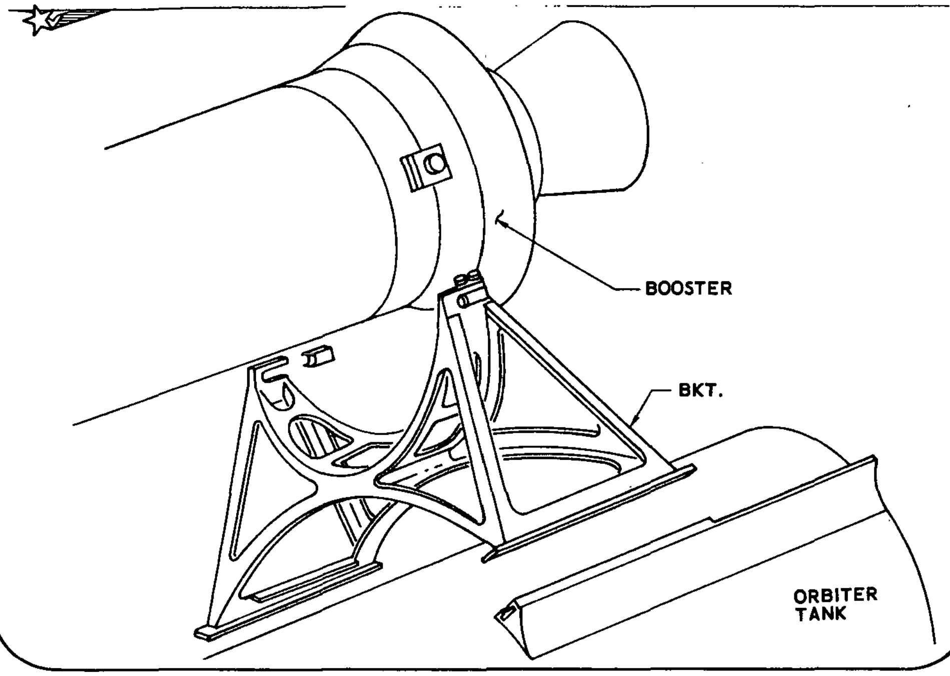


FORWARD END STAGE SUPPORT OF BOOSTER, PARALLEL BURN

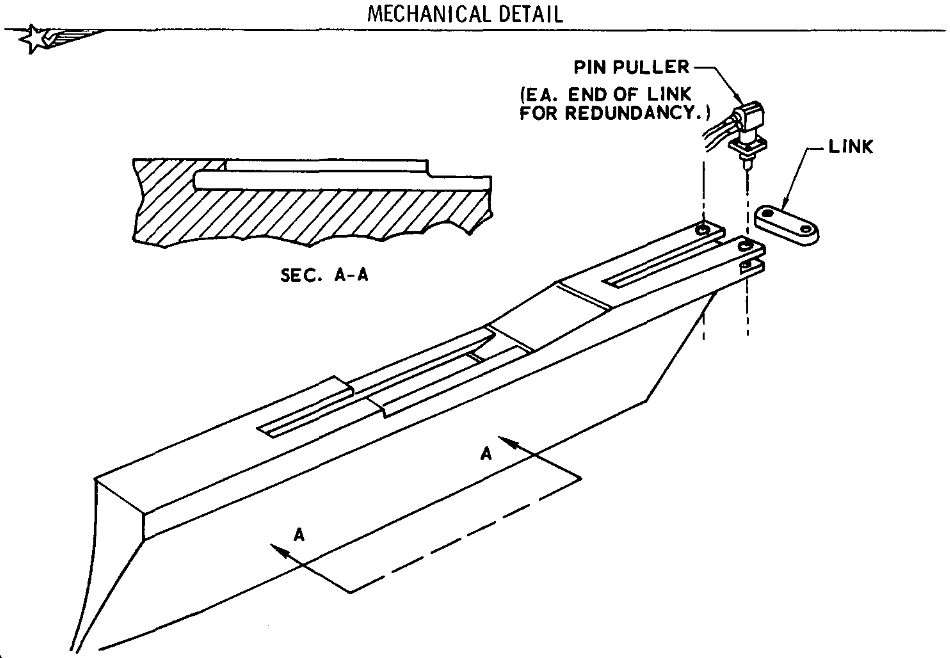
MECHANICAL DETAIL



STAGE AFT SUPPORT OF BOOSTER, PARALLEL BURN
MECHANICAL DETAIL



FORWARD END STAGE TANK ATTACHMENT, PARALLEL BURN
MECHANICAL DETAIL



STAGE MASS PROPERTIES, PARALLEL BURN

NOSE CONE	670 LB
ELECTRICAL	195
EQUIPMENT STRUCTURE	1, 641
PYROTECHNIC	53
AFT HARDWARE	10, 591
STRUCTURE - BOOSTER ATTACH	19, 350
TOTAL	32, 500 LB PER SRM

MECHANICAL DESIGN - PARALLEL BURN
STAGE WEIGHT/CONCLUSIONS

TOTAL WEIGHT OF STAGE COMPONENTS 65, 000 LB

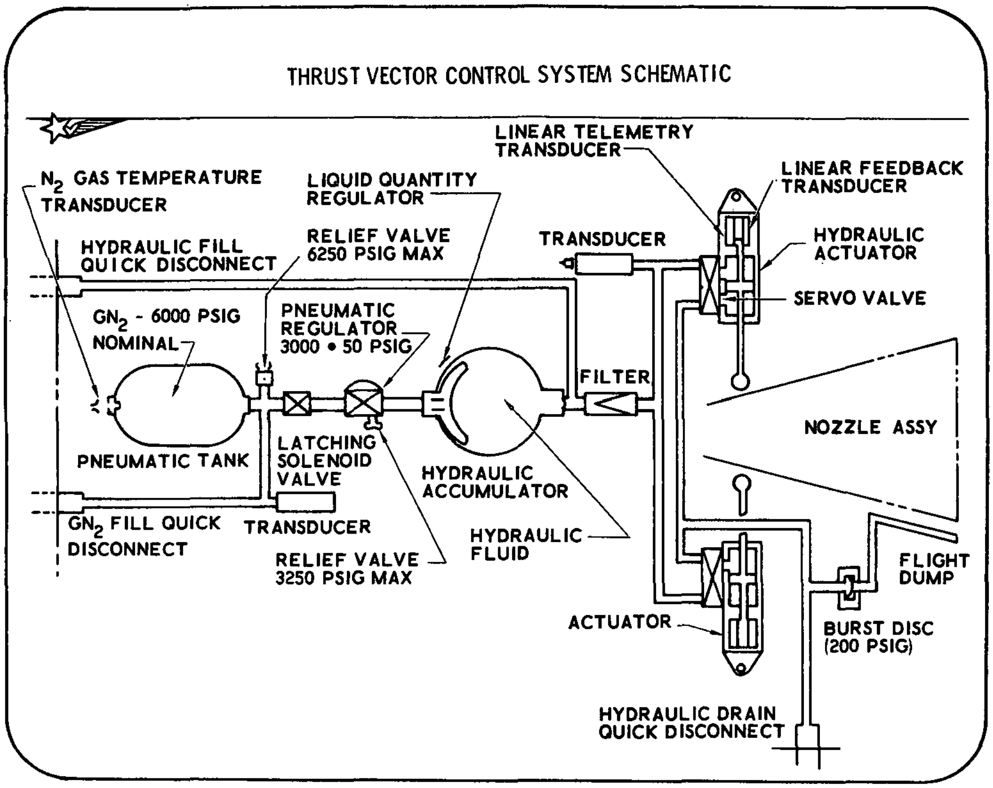
FORWARD SECTION TOTAL	21, 909 LB	AFT SECTION TOTAL	10, 591 LB
NOSE CONE	670		
ELECTRICAL	195		
EQUIPMENT STRUCTURE	1, 641		
PYROTECHNICS	53		
ATTACH STRUCTURE	19, 350	SRM TOTAL	32, 500

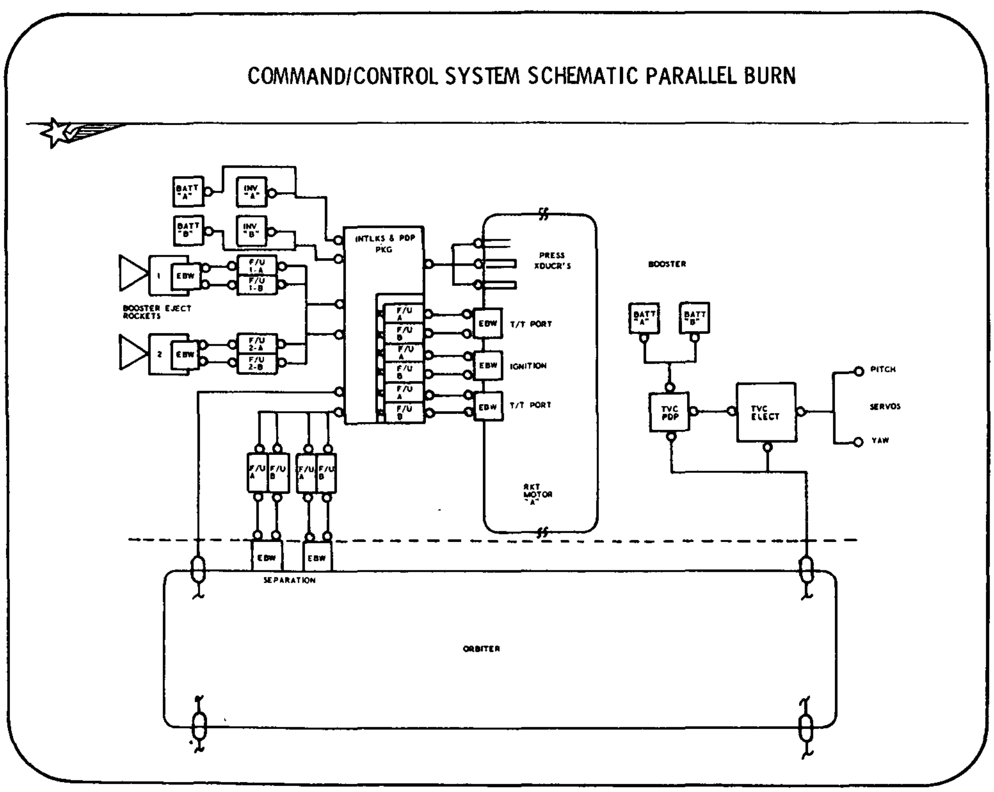
SLIDING RELEASE MUST PROVIDE INHERENT SEPARATION RELIABILITY
- TOLERANCE CONTROL
- FRICTION FORCE
- SEPARATION ROCKET FORCE AND TIMING

CONVENTIONAL DESIGN

NO UNUSUAL PROBLEMS

4.1.3 Flight Control System





4.1.4 Flight Sequencing

PARALLEL BURN LAUNCH/FLIGHT SEQUENCE OF EVENTS

PROPOSED GROUND RULES

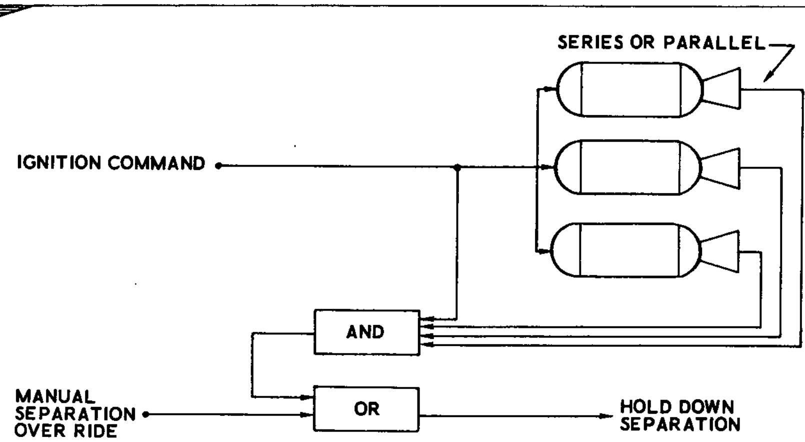
SKIRT CLAMPING IS REQUIRED DURING ORBITER THRUST BUILDUP AND UNTIL SRM IGNITION VERIFIED

BOTH MOTORS WILL BE JETTISONED SIMULTANEOUSLY, ONLY AFTER THRUST LEVEL OF BOTH HAS DROPPED TO ESSENTIALLY ZERO THRUST

THRUST VECTOR CONTROL IS PERFORMED BY SRM ONLY DURING BOOST PHASE

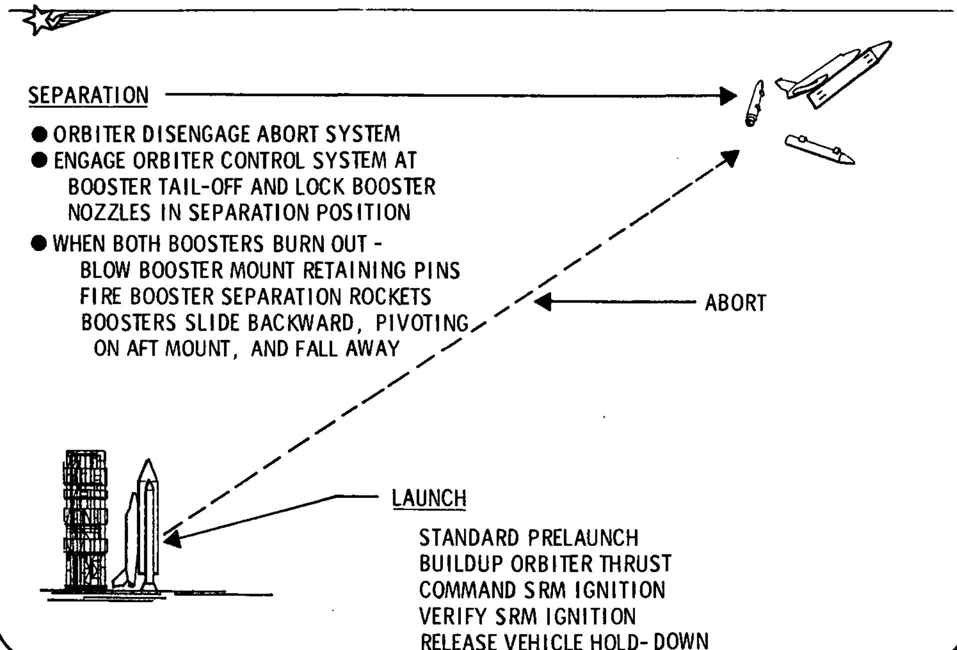
THRUST TERMINATION IS USED ONLY IN THE ABORT MODE.

LAUNCH HOLD DOWN AND IGNITION LOGIC



- IF ANY MOTOR DOES NOT IGNITE
  - HOLD DOWN NOT RELEASED
  - ORBITER COMMANDS SRM THRUST TERMINATION
  - ORBITER TURNS OFF LIQUID ENGINES (PARALLEL)
  - PROCEED THROUGH GROUND ABORT PROCEDURE

PARALLEL BURN LAUNCH/FLIGHT SEQUENCE OF EVENTS



4.2 ALTERNATE DESIGN: 156-INCH SERIES BURN BOOSTER VEHICLE

This subsection contains a description of the tradeoff considerations involved in the selection of electrical systems for the ignition, TVC, and instrumentation components of the alternate series-burn design. Electromechanical, avionics, and related interface systems are then presented in narrative, chart, and schematic form, together with supporting component selection rationale. The flight control system is described schematically next and followed finally by charts detailing the proposed launch/flight sequence.

4.2.1 Tradeoffs

CANDIDATES/TRADEOFFS

SERIES

IGNITION SYSTEM	LOW VOLTAGE *HIGH VOLTAGE CDF/TBI	HIGH VOLTAGE SAFE ELECTRICAL "ARM CONTROL" AND "DISARM" REMOVAL OF INVERTER HIGH VOLTAGE IS LESS SUSCEPTIBLE TO EMI/RFI AND TAMPERING OR SHORTS LOW VOLTAGE LESS EXPENSIVE, BUT REQUIRES INDIVIDUAL MECHANICAL SAFE/ARMS CDF TO BE USED ONLY IF SIMULTANEOUS MULTIPLE OUTPUTS FROM ONE FIRING UNIT REQUIRED
TVC ELECTRICAL SYSTEM	*D.C. ANALOG DIGITAL	DIGITAL COMPATIBLE WITH COMPUTER LOGIC FOR REDUNDANT "VOTING" TECHNIQUES ANALOG IS LESS EXPENSIVE AND HAS SIMPLER INTERFACE WITH ORBITER
INSTRUMENTATION	PULSE CODE MODULATION	PCM HAS HIGHER DATA RESOLUTION
	*PULSE AMPLITUDE MODULATION	PAM IS LESS EXPENSIVE AND PROVIDES ACCEPTABLE DATA.

*BASELINE SELECTION

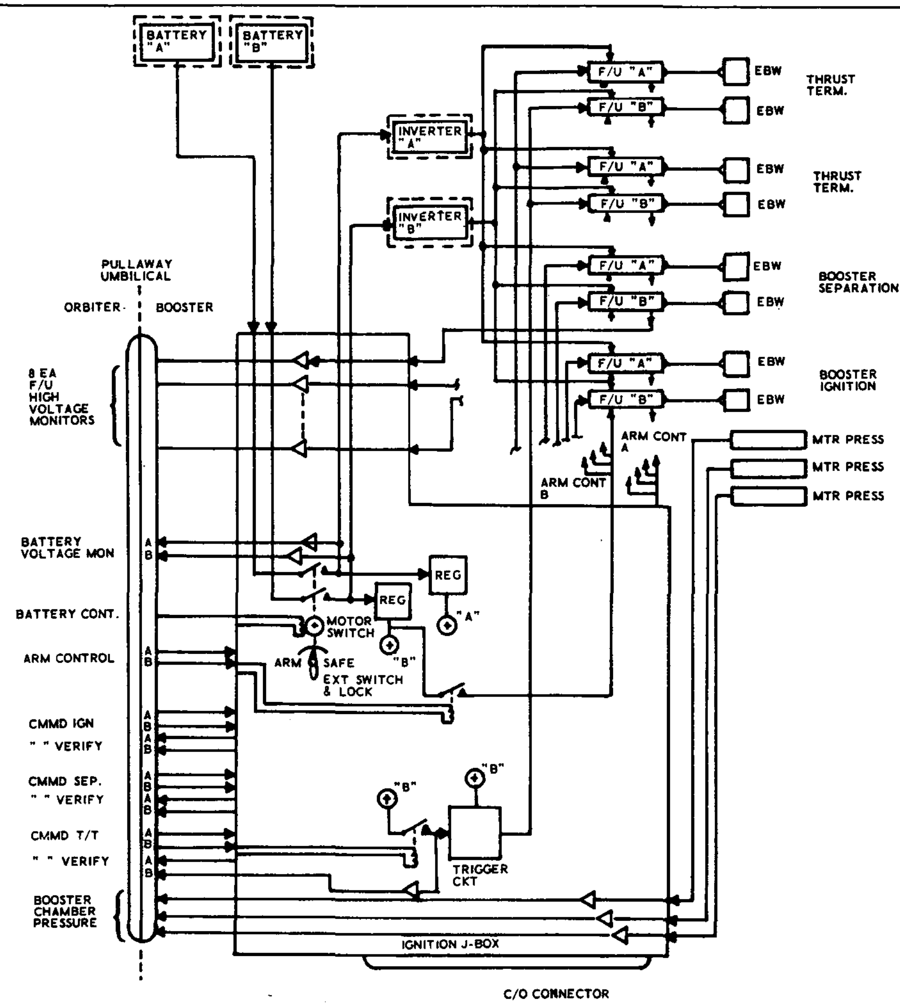
4.2.2 Electromechanical, Avionics, and Related Interfaces

Ignition electrical system. The ignition system is similar to that for the parallel-burn vehicle, except that additional electronics must be added in the forward adapter. The ignition portion of this system will also be dual redundant. It will be used to provide the separation commands for the two separation planes. It will also act as a J-Box to distribute the signals from the orbiter to the three booster motors. It is also necessary to take the orbiter pitch yaw and roll commands and split them between the boosters. The gains will not necessarily be the same for each booster, nor will the polarity. This is the function of the TVC control package.

STAGE FORWARD END ELECTRICAL SYSTEM

- REDUNDANT CIRCUITRY AND POWER SUPPLIES THROUGHOUT IGNITION SYSTEM
- ENTIRE SYSTEM QUALIFIED FOR RANGE AND HUMAN SAFETY
- ELECTRICAL ISOLATION FROM ORBITER AND OTHER SRM CIRCUITS
- A3 TYPE INVERTERS AND FIRING UNITS
- ALL MONITOR CIRCUITS PRECONDITIONED, IMPEDANCE ISOLATION, 0 - +5 VOLT SIGNAL CONDITIONED
- 30 DAY WETSTAND BATTERIES (REMOVABLE)
- INVERTERS REMOVED TO PROVIDE "SAFE" FUNCTION
- EXTERNAL BATTERY "ARM/SAFE" SWITCH
- ELECTRICAL "ARM CONTROL" PROVIDES REMOTE DISARM CAPABILITY FOR IGNITION SYSTEMS
- SINGLE POINT GROUND
- EBW HIGH VOLTAGE IGNITION SYSTEM HAS HIGH SAFETY AND RELIABILITY AS PROVEN ON POLARIS/POSEIDON VEHICLES

STAGE FORWARD END ELECTRICAL SYSTEM SE (SERIES BURN)



<u>TVC electrical system.</u> This is a schematic of the TVC system. Voltage regulators are redundant, with a failure detection logic to switch from the primary to secondary unit in case of failure. Each actuator also has dual feedbacks with a failure detection logic. Triple redundant "majority voting" is used for the servo amplifiers. The input command polarity will be switched, depending on which side of the orbiter the particular booster is located. Though not shown, all input commands from the orbiter are expected to be dual.

TVC ELECTRICAL SYSTEM

GENERAL CHARACTERISTICS

WET CELL BATTERIES (30 DAY WETSTAND)

REDUNDANCY, MAJORITY VOTING AND FAILURE DETECTION IN TVC SYSTEM

TRIPLE VOTING REDUNDANCY ON SERVO AMPLIFIERS

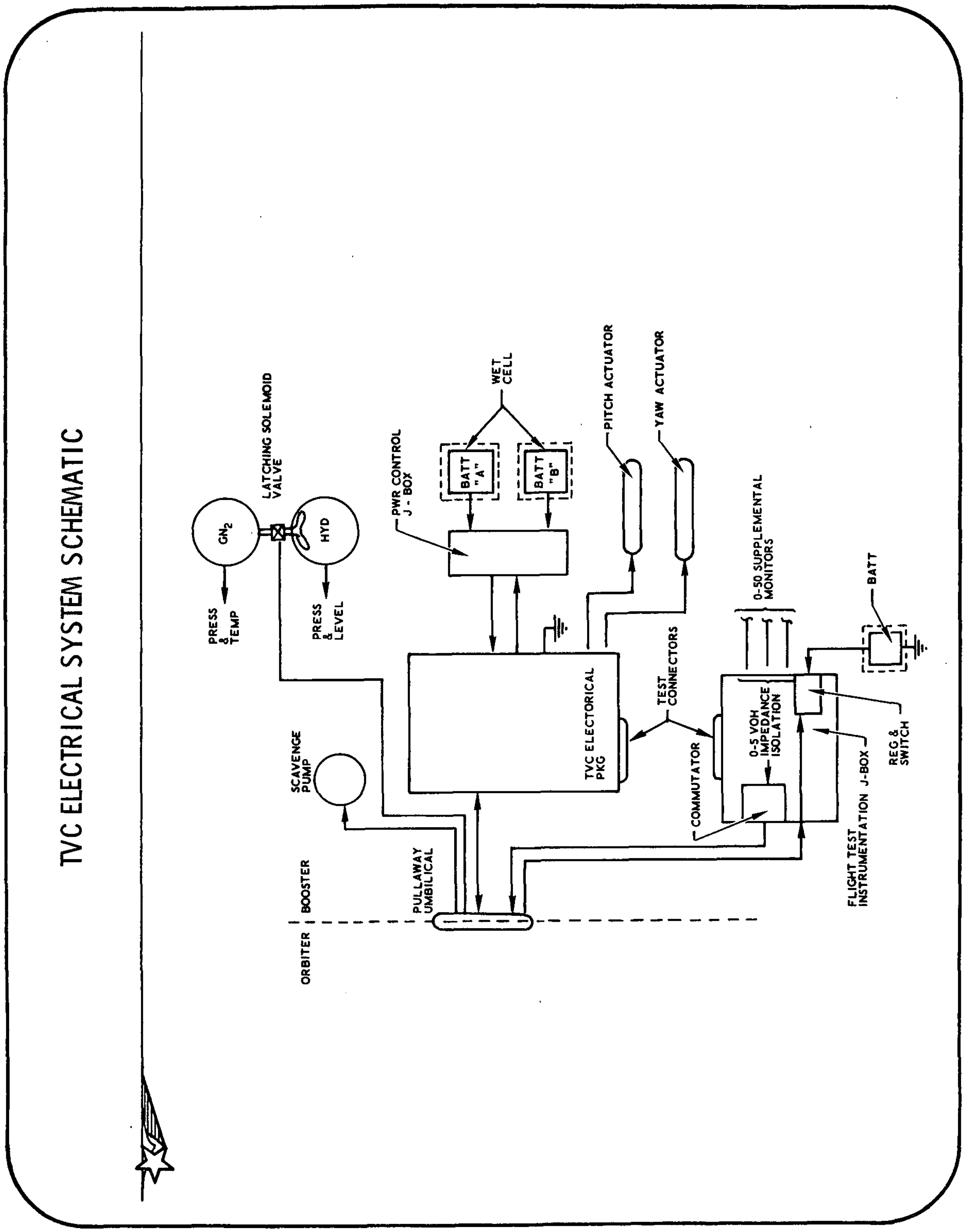
SINGLE POINT GROUND

ELECTRICAL ISOLATION FROM ORBITER AND OTHER SRM CIRCUITS

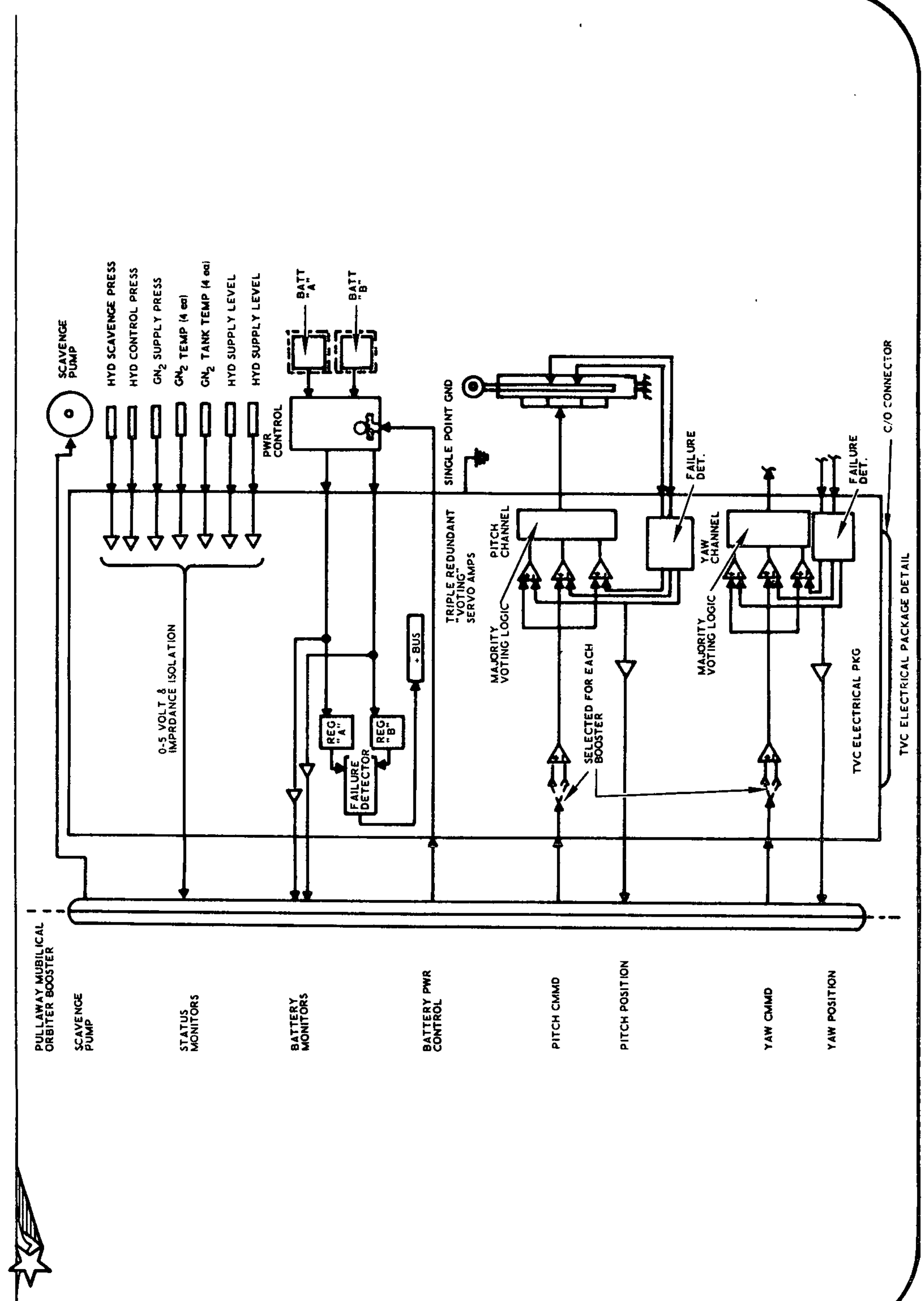
ALL MONITORS CONDITIONED TO 0→5 VOLTS AND IMPEDANCE ISOLATED

DIRECT BYPASS TO RUN OFF FACILITY HYDRAULIC SOURCE

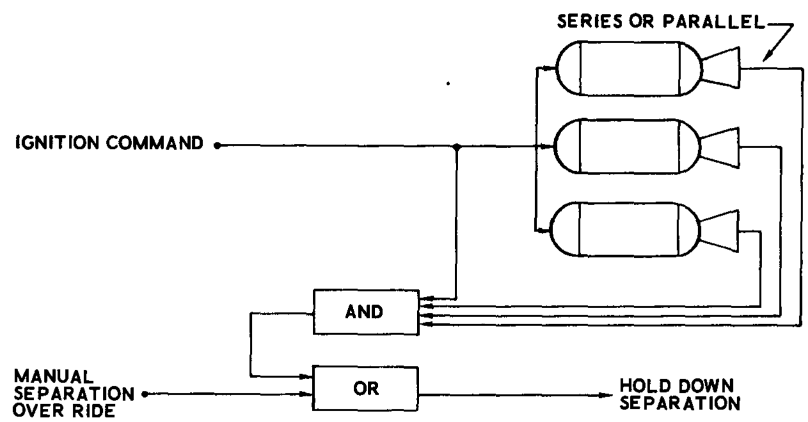
FLIGHT TEST INSTRUMENTATION J-BOX ON INITIAL FLIGHTS ONLY


TVC ELECTRICAL SYSTEM SCHEMATIC
LATCHING SOLEMOID VALVE
GN2
HYD
PRESS & TEMP
PRESS & LEVEL
PWR CONTROL J - BOX
WET CELL
BATT "A"
BATT "B"
PITCH ACTUATOR
YAW ACTUATOR
TVC ELECTORICAL PKG
TEST CONNECTORS
0-50 SUPPLEMENTAL MONITORS
BATT
0-5 VOH IMPEDANCE ISOLATION
REG & SWITCH
COMMUTATOR
FLIGHT TEST INSTRUMENTATION J-BOX
SCAVENGE PUMP
PULLAWAY UMBILICAL
ORBITER
BOOSTER

TVC ELECTRICAL PACKAGE DETAIL



LAUNCH HOLD DOWN AND IGNITION LOGIC



- IF ANY MOTOR DOES NOT IGNITE
  - HOLD DOWN NOT RELEASED
  - ORBITER COMMANDS SRM THRUST TERMINATION
  - ORBITER TURNS OFF LIQUID ENGINES (PARALLEL)
  - PROCEED THROUGH GROUND ABORT PROCEDURE

SERIES BURN STAGE MECHANICAL DESIGN
CANDIDATES AND ALTERNATIVES

INTERSTAGE	LENGTH	DRIVEN BY THRUST NEUTRALIZATION CLEARANCE
	CONFIGURATION - CONICAL VERSUS CIRCLE/TULIP TRANSITION.	CIRCLE/TULIP TRANSITION - LOWER DRAG, STRUCTURAL TIE
	MOTOR ORIENTATION - SINGLE MOTOR INBOARD VERSUS DUAL MOTOR INBOARD	SINGLE MOTOR INBOARD IMPOSES LIGHTER THRUST TERMINATION ENVIRONMENT TO ORBITER
AFT SKIRT	ARRANGEMENT - SINGLE WRAP-AROUND VERSUS THREE INDIVIDUAL	SINGLE WRAP-AROUND PROVIDES SINGLE INTERFACE PLANE WITH LAUNCH PAD. LIGHTER.
SEPARATION JOINTS	LOCATION - ONE LOCATION VERSUS TWO	TWO LOCATIONS PRECLUDE TIP-OFF DAMAGE.

SERIES BURN STAGE MECHANICAL DESIGN
STAGE WEIGHT/SUMMARY CONCLUSIONS

TOTAL STAGE WEIGHT	100,000 LB		
INTERSTAGE TOTAL	76,243 LB	AFT TOTAL	23,757
ELECTRICAL	174 LB	ELECTRICAL	946
PYROTECHNIC	34 LB	STRUCTURE	22,811
STRUCTURE	76,035 LB		

THRUST NEUTRALIZATION CLEARANCE REQUIREMENTS ARE DRIVING INTERSTAGE LENGTH, WHICH INFLUENCES BENDING MOMENTS, AND DYNAMIC FREQUENCIES, NEEDS ADDITIONAL STUDY.

SERIES BURN MECHANICAL DESIGN FEATURES

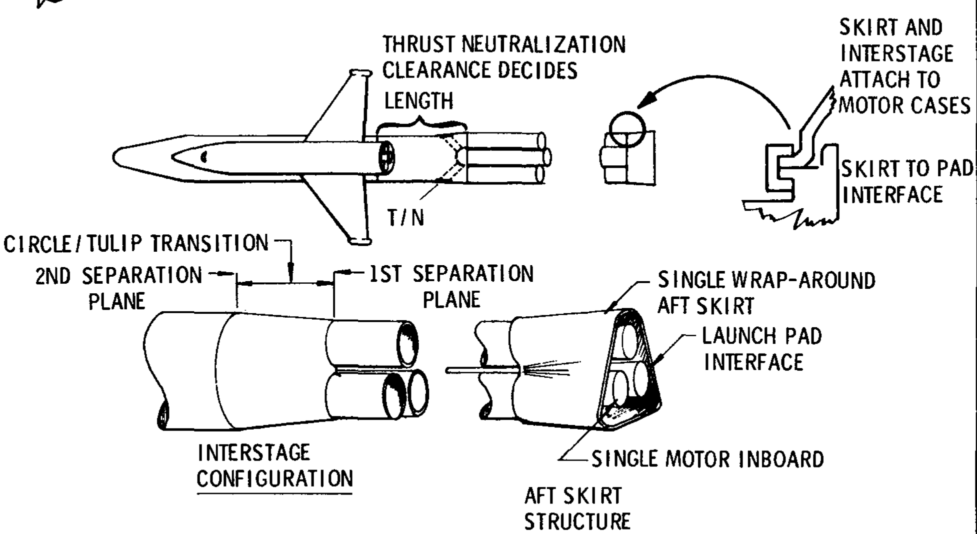
CONTOURED TRANSITION IN INTERSTAGE AND AFT SKIRT REDUCES DRAG, FACILITATES CLUSTERED MOTOR ATTACHMENT

INTERSTAGE LENGTH DETERMINED BY THRUST TERMINATION CLEARANCE REQUIREMENT ON ORBITER

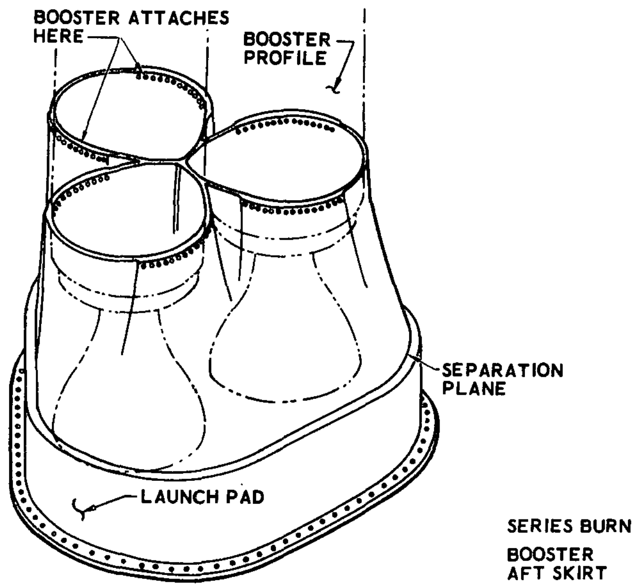
SINGLE WRAP-AROUND FLARED SKIRT SHIELDS ALL NOZZLES, TAKES PAD VEHICLE SUPPORT AND HOLD DOWN LOADS

CONDUIT DOWN CLUSTER CENTER ELIMINATES NEED FOR MOTOR RACEWAYS

SERIES BURN STAGE MECHANICAL DESIGN



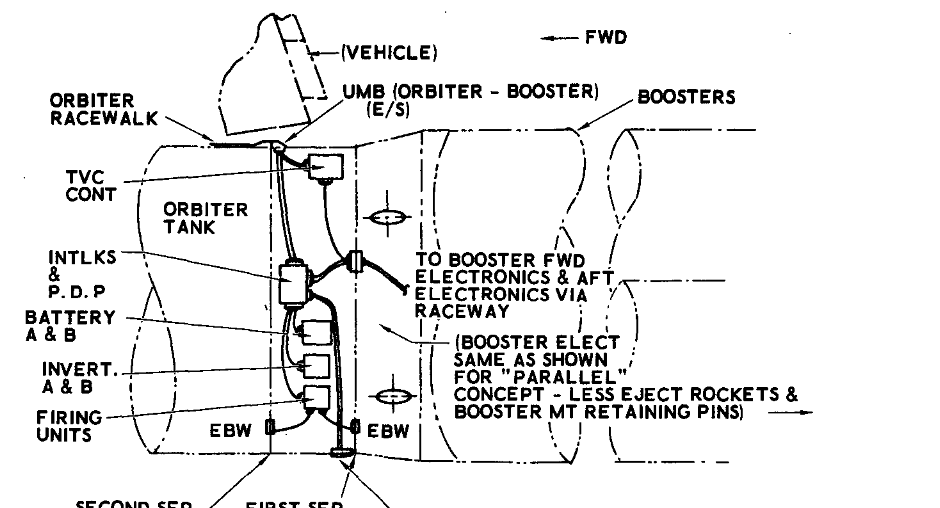
SERIES BURN BOOSTER AFT SKIRT



SERIES BURN MASS PROPERTIES

INTERSTAGE ELECTRICAL	174
INTERSTAGE PYROTECHNIC	34
INTERSTAGE STRUCTURE	76, 035
AFT STRUCTURE	22, 811
AFT STRUCTURE ELECTRICAL	946
TOTAL	100, 000

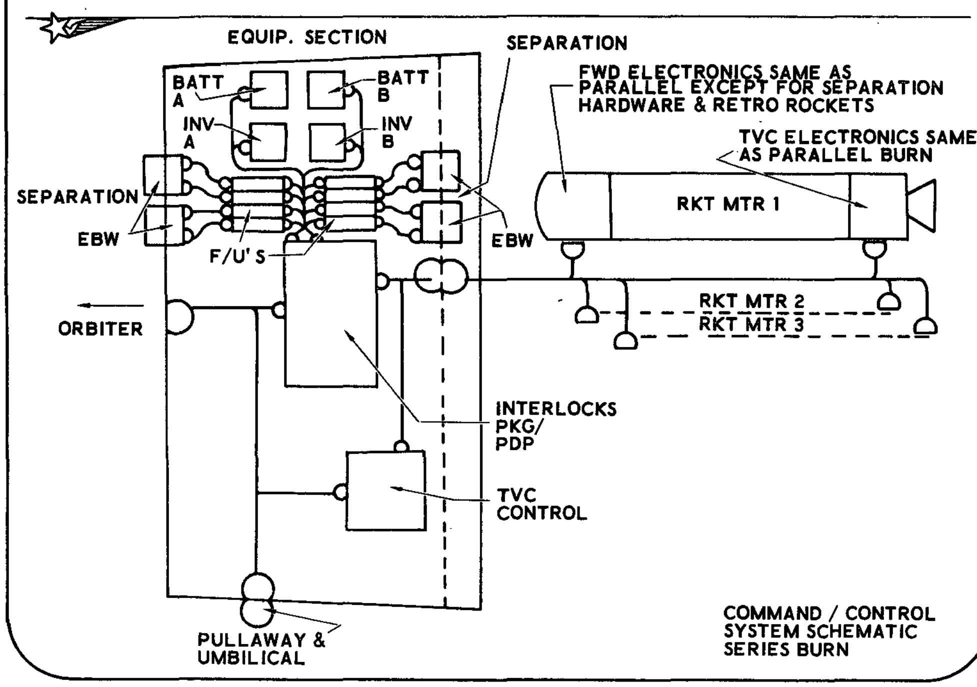
SERIES BURN ELECTRICAL/ORDNANCE



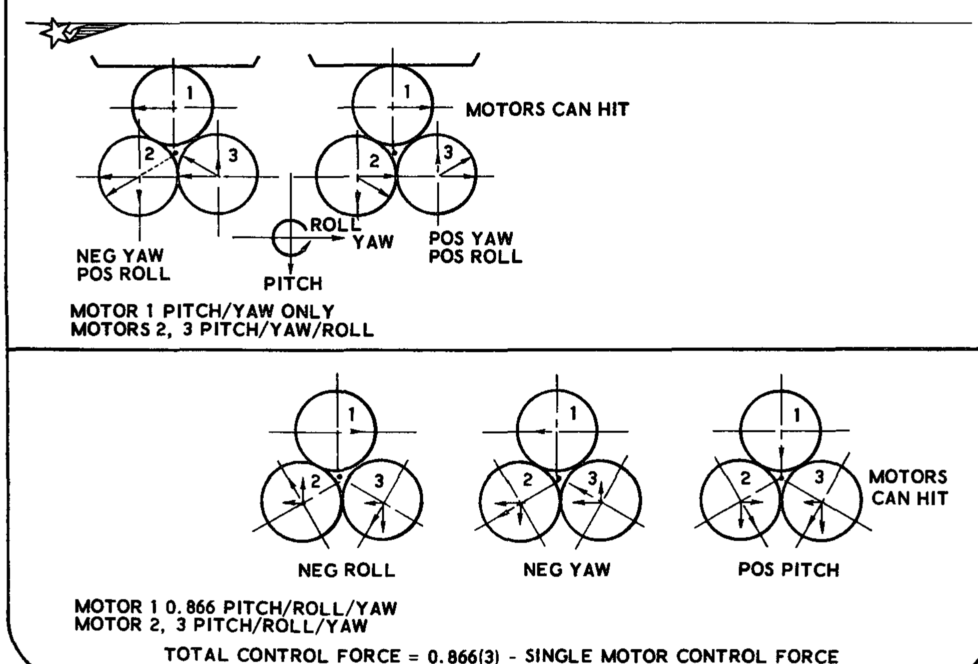
FEATURES: HIGH VOLTAGE EBW INITIATION SYSTEMS FOR SAFETY
REDUNDANT CIRCUITRY AND POWER SUPPLIES FOR RELIABILITY
CONDUIT THROUGH CLUSTER, NO RACEWAY

4.2.3 Flight Control System

COMMAND/CONTROL SYSTEM SCHEMATIC SERIES BURN



SERIES BURN TVC CONTROL SCHEMATICS



4.2.4 Flight Sequencing

FLIGHT SEQUENCING

PROPOSED GROUND RULES

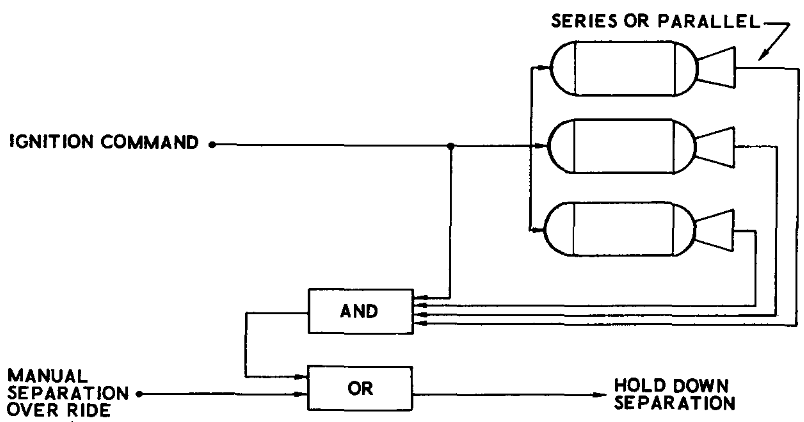
SKIRT CLAMPING IS MAINTAINED UNTIL IGNITION VERIFICATION OF ALL SRM's

ALL THREE SRM's WILL BE JETTISONED SIMULTANEOUSLY AS CLUSTERED UNIT

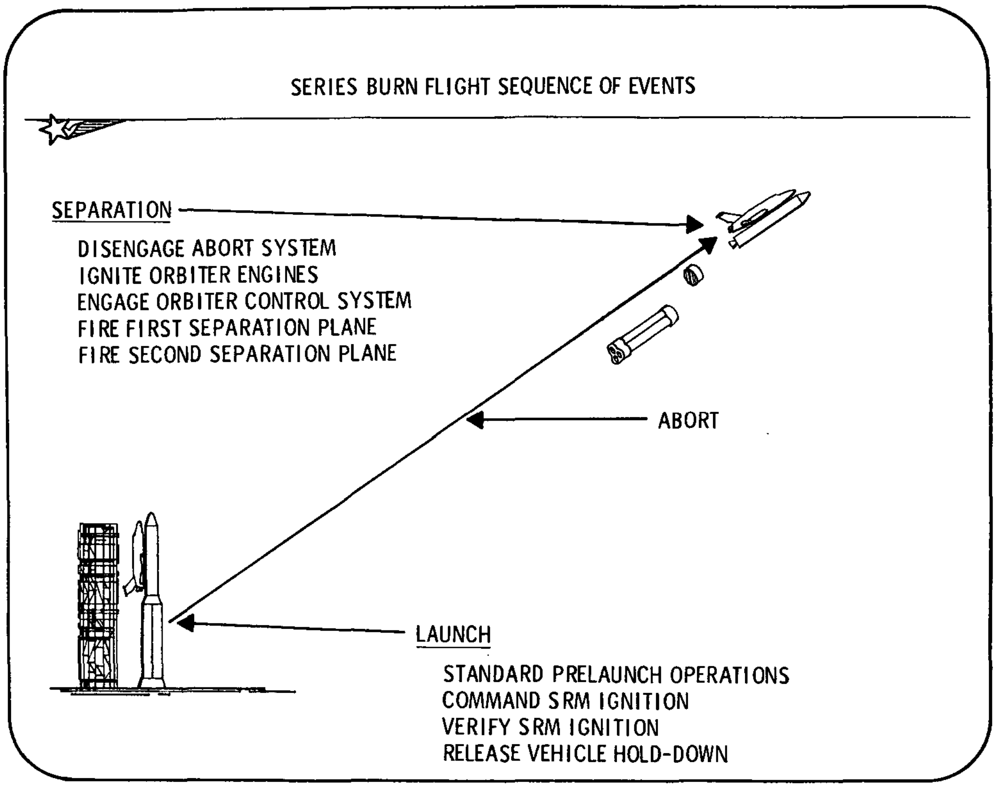
FORWARD PORTION OF INTERSTAGE IS JETTISONED AFTER THE SRM CLUSTER IS JETTISONED

THRUST TERMINATION IS USED ONLY IN THE ABORT MODE

LAUNCH HOLD DOWN AND IGNITION LOGIC



- IF ANY MOTOR DOES NOT IGNITE
  - HOLD DOWN NOT RELEASED
  - ORBITER COMMANDS SRM THRUST TERMINATION
  - ORBITER TURNS OFF LIQUID ENGINES (PARALLEL)
  - PROCEED THROUGH GROUND ABORT PROCEDURE


SERIES BURN FLIGHT SEQUENCE OF EVENTS
SEPARATION
DISENGAGE ABORT SYSTEM
IGNITE ORBITER ENGINES
ENGAGE ORBITER CONTROL SYSTEM
FIRE FIRST SEPARATION PLANE
FIRE SECOND SEPARATION PLANE
ABORT
LAUNCH
STANDARD PRELAUNCH OPERATIONS
COMMAND SRM IGNITION
VERIFY SRM IGNITION
RELEASE VEHICLE HOLD-DOWN

Section 5

SYSTEM DEFINITION

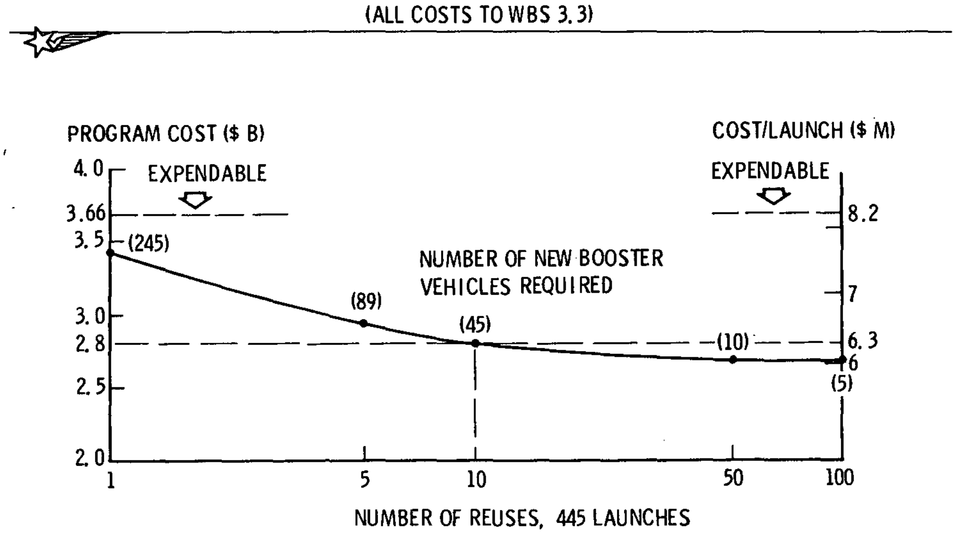
5.1 KEY ISSUES

As part of the study program, LPC evaluated the key issues related to the SRM booster: recovery/reuse, abort, and ecological considerations.

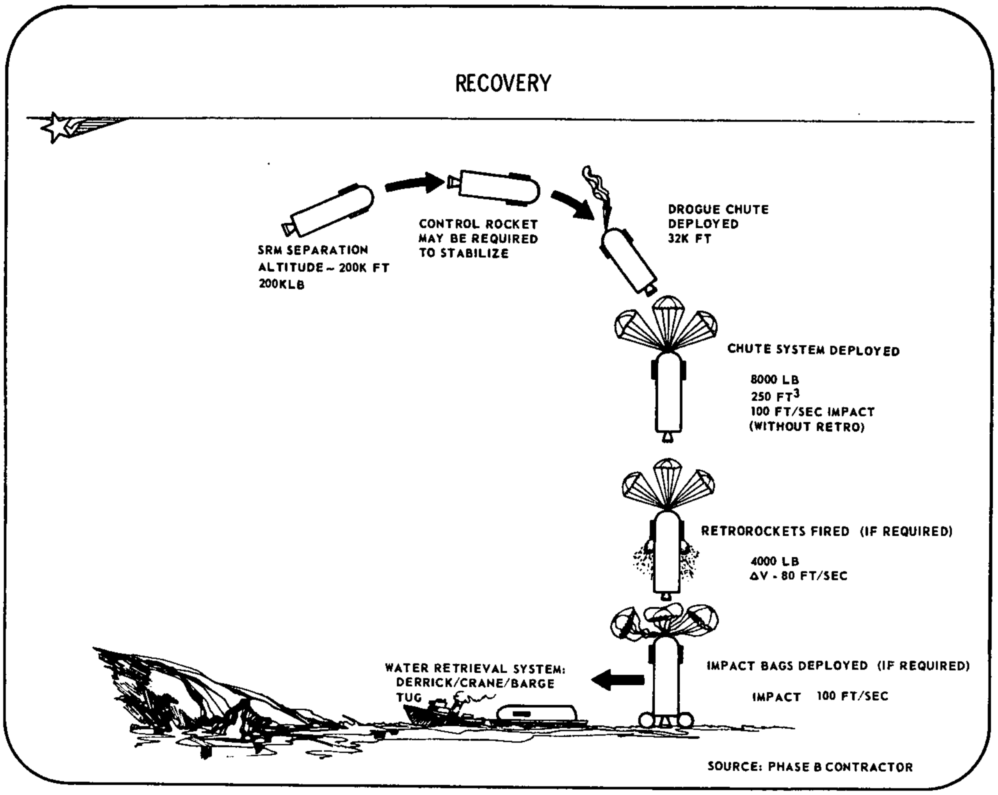
5.1.1 Recovery/Reuse of SRM Booster

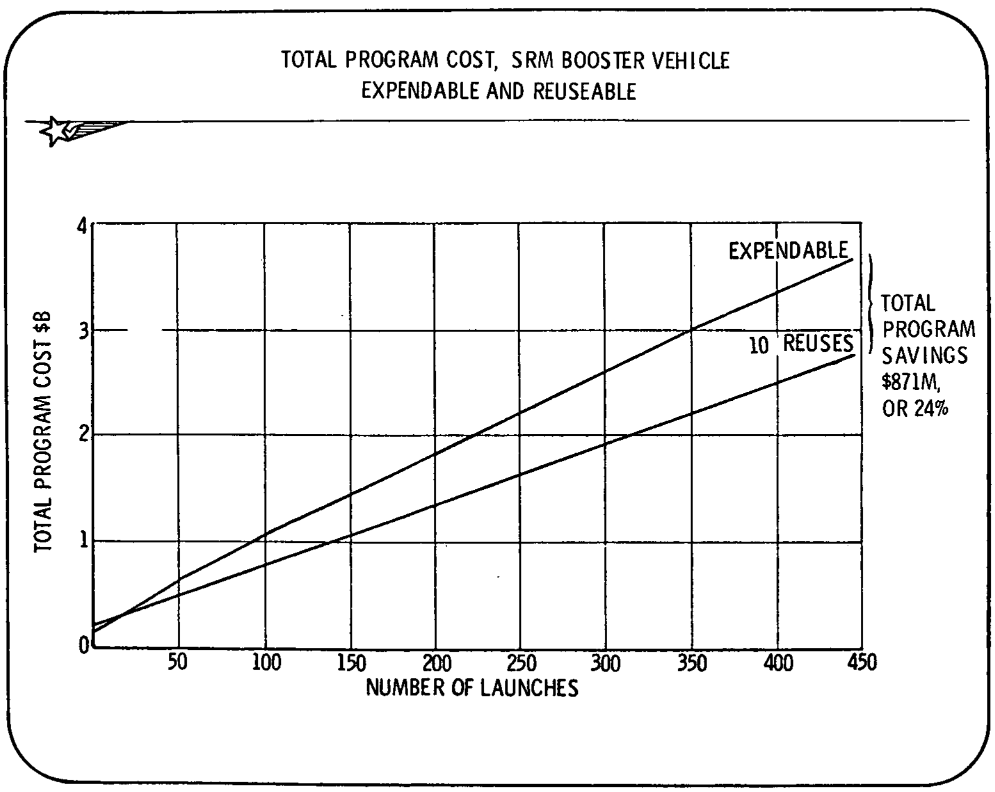
The following charts present the results of LPC's study of the recovery and reuse of the SRM booster.

EFFECT OF RECOVERY/REUSE
TOTAL PROGRAM COSTS & LAUNCH COSTS
(ALL COSTS TO WBS 3.3)



NOTE: MOST OF THE SAVINGS AVAILABLE BY RECOVERY / REUSE ARE ACHIEVED WITH 10 REUSES - ALL REPORTED COSTS FOR RECOVERY BASED ON 10 REUSES


RECOVERY
SRM SEPARATION
ALTITUDE ~ 200K FT
200KLB
CONTROL ROCKET
MAY BE REQUIRED
TO STABILIZE
DROGUE CHUTE
DEPLOYED
32K FT
CHUTE SYSTEM DEPLOYED
8000 LB
250 FT3
100 FT/SEC IMPACT
(WITHOUT RETRO)
RETROROCKETS FIRED (IF REQUIRED)
4000 LB
ΔV - 80 FT/SEC
IMPACT BAGS DEPLOYED (IF REQUIRED)
IMPACT 100 FT/SEC
WATER RETRIEVAL SYSTEM:
DERRICK/CRANE/BARGE
TUG
SOURCE: PHASE B CONTRACTOR


TOTAL PROGRAM COST, SRM BOOSTER VEHICLE
EXPENDABLE AND REUSEABLE
TOTAL PROGRAM COST $B
0
1
2
3
4
50
100
150
200
250
300
350
400
450
NUMBER OF LAUNCHES
EXPENDABLE
10 REUSES
TOTAL
PROGRAM
SAVINGS
$871M,
OR 24%

RECOVERY REFURBISHMENT / REUSE
COST SUMMARY - 156-INCH PARALLEL BURN

	TOTAL PROGRAM SAVINGS			
	100 REUSES	50 REUSES	10 REUSES	5 REUSES
	($M)			
SRM	853	822	766	686
STAGE ATTACH STRUCTURE AND RECOVERY EQUIP.*	136	143	105	57
TOTAL	989	975	871	743
SAVINGS / LAUNCH				
26M DEVELOPMENT	2.2	2.2	2.0	1.7
100M DEVELOPMENT	2.1	2.0	1.8	1.5
TOTAL COST / LAUNCH - EXPENDABLE $8.2M				

*INCLUDES EXPENSE FOR CHUTES, RECOVERY, RETRIEVAL, AND REFURBISHMENT DEVELOPMENT AND FACILITIES

SRM BOOSTER VEHICLE
TOTAL PROGRAM COST OF MAJOR ELEMENTS / LAUNCH ($M)
(EXPENDABLE AND RECOVERABLE)

MAJOR ELEMENT	EXPENDABLE	1 REUSE	5 REUSES	10 REUSES	50 REUSES	100 REUSES
SRM	6.0	5.4	4.6	4.4	4.2	4.2
STAGE AND LAUNCH	2.2	1.7	1.6	1.5	1.5	1.4
RECOVERY	—	0.5	0.4	0.4	0.4	0.4
RETRIEVAL	—	0.04	0.04	0.04	0.04	0.04
TOTAL COST / LAUNCH	8.2	7.6	6.6	6.3	6.1	6.0

LPC RECOVERY/REUSE GROUNDRULES

REUSEABLE COMPONENTS

- MOTOR CHAMBER
- IGNITER CASE
- NOZZLE METAL PARTS
- TVC ACTUATORS
- TVC POWER SYSTEM
- LOCKSEAL METAL PARTS
- STAGE ATTACH STRUCTURE
- RECOVERY CHUTES

VALUE OF REUSEABLE COMPONENTS: \$3.256M/LAUNCH - AVERAGE COST FOR 445 LAUNCH QUANTITY - NO ESCALATION INCLUDED FOR REDUCED BUY QUANTITIES

AVERAGE COST TO REFURBISH: 10 PERCENT OF ORIGINAL COST, OR \$0.326M/LAUNCH

DEVELOPMENT COST OF: RECOVERY SYSTEM \$14M - SOURCE: LMSC
RETRIEVAL (WATER SYSTEM) \$5M - SOURCE: LMSC/NASA

FACILITY COST FOR REFURBISHMENT: \$6M

RECURRING COST OF: RECOVERY \$0.249M/LAUNCH - SOURCE: LMSC
RETRIEVAL \$0.034M/LAUNCH - SOURCE: LMSC

COMPONENT LOSS RATE PRIOR TO REFURBISHMENT: 10 PERCENT

SRM BOOSTER VEHICLE
EXPENDABLE VS REUSABLE DESIGN CONSIDERATIONS

CONCLUSIONS: NO CHANGE IN SRM DESIGN - EXPENDABLE VERSUS REUSABLE

CHAMBER PRESSURE SAFETY FACTOR: 1.4 $\left(\frac{\text{BURST PRESSURE}}{\text{MEOP}}\right)$ NECESSARY FOR MAN-RATED SYSTEM; ADEQUATE FOR MULTIPLE REUSE.

CHAMBER STRUCTURE IMPACT RESISTANCE: EVALUATION INDICATES ADEQUATE FOR VERTICLE $\pm 45^{\circ}$ WATER ENTRY, EITHER NOSE OR NOZZLE DOWN.

REUSABLE COMPONENT MATERIALS: BASELINE SELECTION SATISFACTORY FOR < 24-HOUR SALT WATER EXPOSURE WITHOUT DEGRADING SAFETY MARGIN.

SRM EXPENDABLE COMPONENTS: NOT AFFECTED.

STAGE ELECTRICAL/AVIONICS CONSIDERED EXPENDABLE.

STAGE ATTACH STRUCTURE: MUST REMAIN WITH SRM DURING SEPARATION AND WATER ENTRY; SAME REQUIREMENTS FOR MATERIALS.

PARACHUTE SYSTEM MUST BE DESIGNED FOR MULTIPLE REUSE FOR MAXIMUM COST SAVINGS

PROGRAM COSTS ASSIGNABLE TO
RECOVERY - REFURBISHMENT POTENTIAL

			$ MILLION
TOTAL EXPENDABLE PROGRAM			3,664
NON-RECOVERABLE EXPENDITURES, BOOSTER VEHICLE			
FACILITIES, DEVELOPMENT, TRANSPORTATION, G&A	943		
STAGE ELECTRICALS, CONTROLS, ASSEMBLY	573		
SRM MATERIALS AND TOOLING	538		
SRM DEVELOPMENT MATERIALS	71		
		2,125	
POTENTIAL RECOVERABLE EXPENDITURES, BOOSTER VEHICLE			
MOTOR CASE	651		
NOZZLE	270		
IGNITER	22		
TVC	168		
LOCKSEAL	62		
STAGE STRUCTURE	366		
		1,539	3,664
NON-RECOVERABLE EXPENDITURES, REUSE SYSTEM			
FACILITIES, DEVELOPMENT	25		
RECURRING ASSEMBLY AND LAUNCH EFFORT	126		
		151	
POTENTIAL RECOVERABLE EXPENDITURE, REUSE SYSTEM			
PARACHUTE SYSTEM	177		
		177	319

COST EFFECT OF VARIOUS REUSE RATES
- BOOSTER VEHICLE -

	($ MILLIONS) NEW SETS / REFURBISH CYCLES / NUMBER OF REUSES				
	5/440/100	10/435/50	45/400/10	89/356/5	245/222/1
NEW UNITS @ 3.813 / LAUNCH	19	38	172	339	934
REFURBISH @ 0.883 / LAUNCH	389	384	354	315	196
FIXED COSTS	2,267	2,267	2,267	2,267	2,267
TOTAL PROGRAM COSTS - REUSABLE	2,675	2,689	2,793	2,921	3,397
TOTAL PROGRAM COSTS - EXPENDABLE	3,664	3,664	3,664	3,664	3,664
TOTAL PROGRAM SAVINGS	989	975	871	743	267
COST/LAUNCH - REUSABLE	6.0	6.1	6.3	6.6	7.6
COST/LAUNCH - EXPENDABLE	8.2	8.2	8.2	8.2	8.2
SAVINGS/LAUNCH	2.2	2.1	1.9	1.6	0.6

COST EFFECT
OF RECOVERY-REFURBISHMENT ON COMPONENTS

REUSABLE COMPONENT	NEW	EXPENDABLES	RECOVERABLES
	COST / LAUNCH ($1K)		
MOTOR CASE	1,440	-0-	1,440
NOZZLE	607	407	200
IGNITER ASSEMBLY	49	24	25
TVC ASSEMBLY	378	16	362
LOCKSEAL	140	70	70
SRM TOTAL	2,614	517	2,097
STAGE ATTACH STRUCTURE	800	20	780
PARACHUTE	399	20	379
BOOSTER VEHICLE TOTAL	3,813	557	3,256
		REPLACE	REFURBISH
COST TO:		557	326 (10%)

REFURBISH, REUSE COST SAVINGS

156-INCH SRM PARALLEL BURN — 445 LAUNCHES

COMPONENT	REUSE VALUE / LAUNCH ($K)	COST SAVINGS / LAUNCH ($K) 50 REUSES	10 REUSES
MOTOR CASE	1,440	1,265	1,165
NOZZLE METAL PARTS	200	176	160
TVC SYSTEM	362	319	289
LOCKSEAL METAL PARTS	70	62	57
IGNITER CASE	25	22	20
STAGE STRUCTURE	780	685	620

Recovery/reuse; prevention of SRM case corrosion. Like all alloy steels, D6AC steel corrodes when exposed to marine environments. In flowing sea water the general corrosion rate of D6AC is only 0.007 inch per year, but pitting can occur and cause localized damage at a faster rate. Tests on D6AC steel at LPC have shown that although general corrosion occurs after minutes, pitting does not occur within 24 hours of exposure to salt water. With the D6AC coated by a system of paints and sealants, the exposure time required to cause pitting is greatly increased. It is therefore feasible to protect the case from degradative corrosion for the period of time required for recovery from the ocean.

The corrosion prevention materials required to protect the SRM case would consist of insulation on the inside, a two-part paint system on the outside, and sealants at all joints. The insulation will be applied in sufficient thickness to prevent both thermal degradation of the steel during motor firing and corrosive damage due to ocean exposure. The paint will be of the epoxy or polyurethane type and will contain chromate inhibitors so that pitting of the steel will be prevented even if the paint is scratched. The sealants used will also contain chromate inhibitors. The clevis joints and pins will be sprayed with sealant and then assembled wet. The exposed seams on the inside and outside of the case joints will be calked with a bead of sealant to prevent water seepage into the joints.

Corrosion prevention materials of the type described are commonly used on Lockheed aircraft of the antisubmarine type, which are subjected to severe corrosive environments for thousands of hours. The materials effectively prevent degradative corrosion in such environments and should certainly be sufficient for the Space Shuttle.

Lockheed Propulsion Company uses materials of the types described to protect the air-carried, D6AC steel SRAM motor case. Tests have verified that the D6AC case can be exposed to more than 50 hours of salt spray, 15 days of 90-percent humidity, and 24 hours of water immersion without pitting occurring. LMSC uses similar paint systems to protect submergence vehicles such as Deep Quest and DSRV. These vehicles have alloy-steel hulls and are subjected to much longer exposure times than will be required for the SRM case. No problems with corrosion have been encountered on these vehicles, even after numerous missions.

This experience verifies that the SRM case can be adequately protected from severe corrosion during the ocean recovery phase of the orbital mission. After the case is retrieved and loaded on a barge, it will be washed with inhibited fresh water and then sprayed with a water-displacing oil so that corrosion will not occur during the trip back to the disassembly facility. In this respect, a solid rocket motor offers definite advantages over a liquid motor. The lower weight of the solid motor case makes speedy recovery and preservation of the case possible. The simpler design of the solid case is also advantageous because of the fewer faying surfaces requiring protection and because of the lack of any dissimilar-metal contact.

It should be noted that if surface corrosion of the steel does occur in some areas because of unusual exposure conditions, such as a delay before retrieval, the case can still be refurbished and reused. The thin layer of corrosion products (rust) can be removed by abrasive methods, and if small pits are present, they can be blended out. Since the nominal wall thickness of the case will be 0.02 inch above the minimum allowable, it is not possible that the surface damage or pits could have sufficient depth to render the case unsuitable for reuse. Since the motor case will not contain welds, environmental flaw growth will not be a problem.

5.1.2 Abort

Abort assessment. The primary objective of an abort operation must be personnel safety. If the only personnel on board were the flight crew, then the incorporation of a crew escape capsule/system in the Orbiter design would be clearly indicated. However, since there may be personnel in the payload compartment of any flight, the escape system must be capable of boosting both the cockpit and the payload compartment to safety. The practical constraints against separating the Orbiter into two compartments dictates that the escape system boost the entire Orbiter to safety.

In the parallel-burn configuration, with Orbiter engines operating, escape can be accomplished either through an on-board hydrogen/oxygen (H/O) supply or by auxiliary solid rockets similar to those of the Apollo system. The latter method is preferred because it can override an Orbiter engine failure. The incorporation of this auxiliary excape system must be evaluated in the Orbiter design phases.

The critical time for abort occurs from lift-off to approximately 30 seconds. During this period, the vehicle has not reached sufficient altitude or velocity, given Orbiter separation, for the Orbiter to maneuver to a safe landing. It is this condition that dictates the use of the Orbiter escape system. The thrust-to-weight ratio (<1) of the Orbiter and its H/O tank is an additional constraint against Orbiter independence.

Until booster burnout, three conditions can cause the need to abort: (1) SRM malfunction, (2) Orbiter engine malfunction, and (3) malfunction of a critical Orbiter system. If the condition arises in the Orbiter and does not endanger safety, then the booster stage should be allowed to burn out in order to provide maximum maneuverability to the Orbiter.

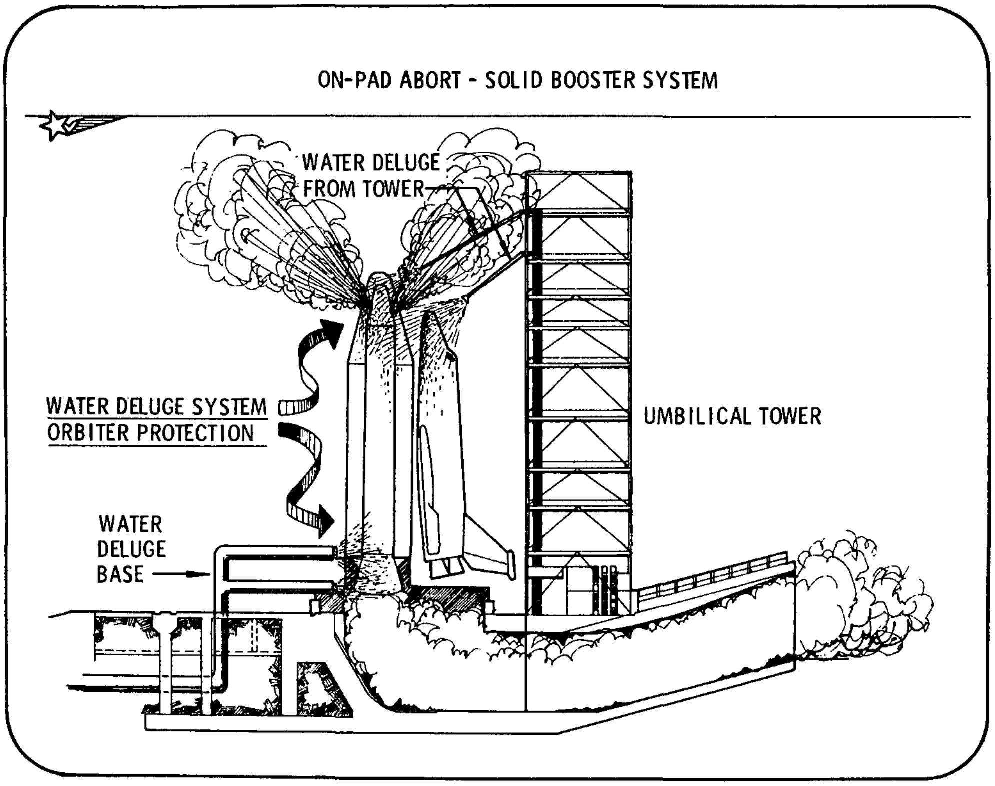
GROUND ABORT PROCEDURE
PRIOR TO SRM IGNITION

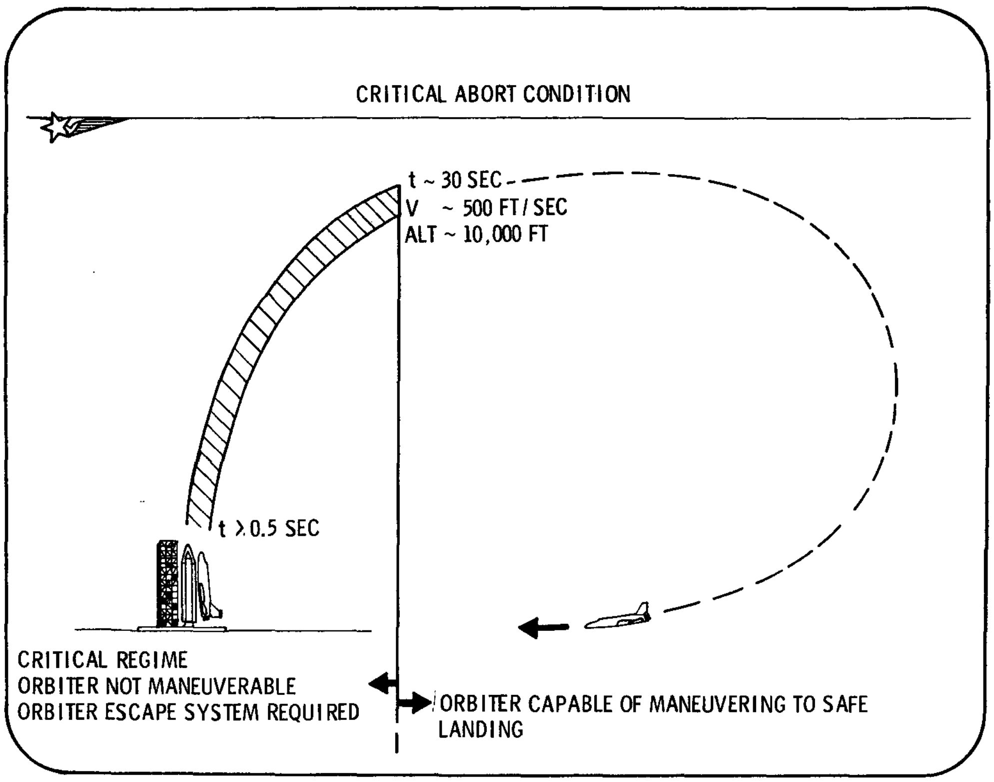
ELECTRICAL SHUT DOWN SEQUENCE

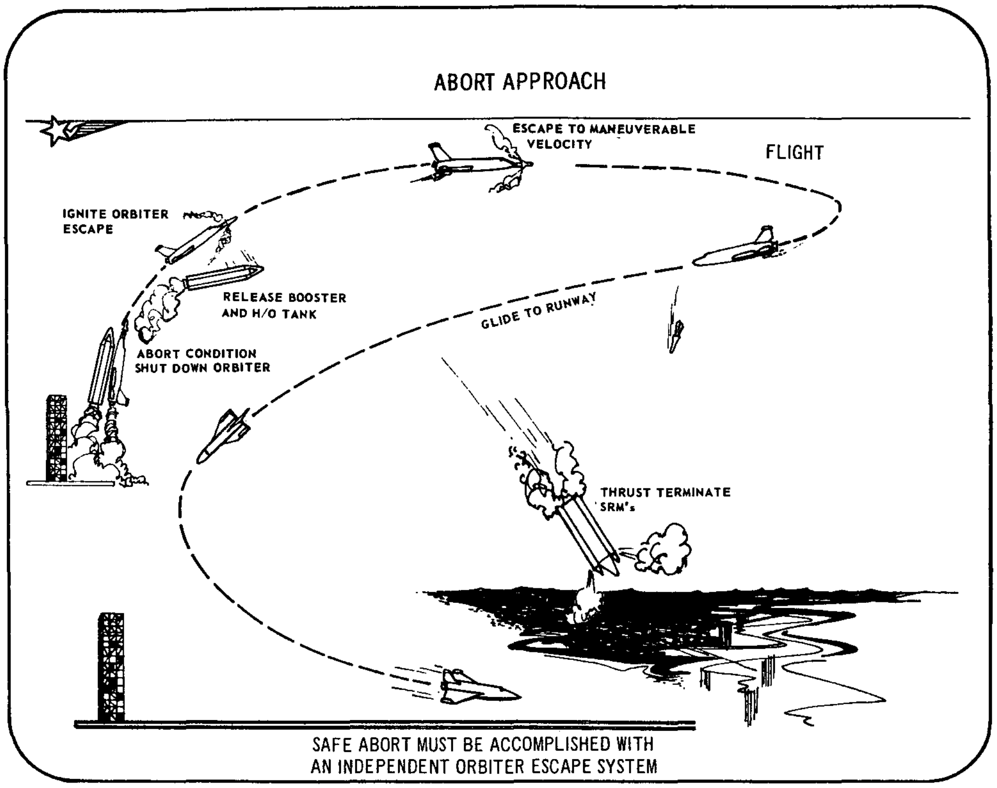
1. REMOVE ARM CONTROL VOLTAGE (VERIFY F/U's DISCHARGE)
2. REMOVE IGNITION AND TVC SYSTEM BATTERY POWER (VERIFY VOLTAGE IS 0)
3. LOCK IGNITION BATTERY TRANSFER SWITCH OFF (ON VEHICLE)
4. REMOVE IGNITION INVERTERS
5. REMOVE BATTERIES (IF REQUIRED)
6. REMOVE PYROTECHNIC HARDWARE (SQUIBS AND RETROROCKETS) (IF REQUIRED)

Abort approach. The abort sequence is shown for two conditions: on-pad and in-flight. The following launch sequence helps define the on-pad condition: (1) ignite Orbiter engines; (2) verify Orbiter thrust; (3) ignite SRMs; (4) verify SRM thrust; (5) release hold-down mechanism. Failure of one SRM to ignite thereby becomes the primary cause for an on-pad abort.

Automatic and manual abort initiation have been treated. For an SRM malfunction, either could be activated as a result of SRM sensor read-out in the Orbiter. Thermal and pressure sensors are planned. If automatic abort provisions are incorporated, voting procedures should be used where possible (e. g., multiple pressure transducers) to avoid inadvertent abort.



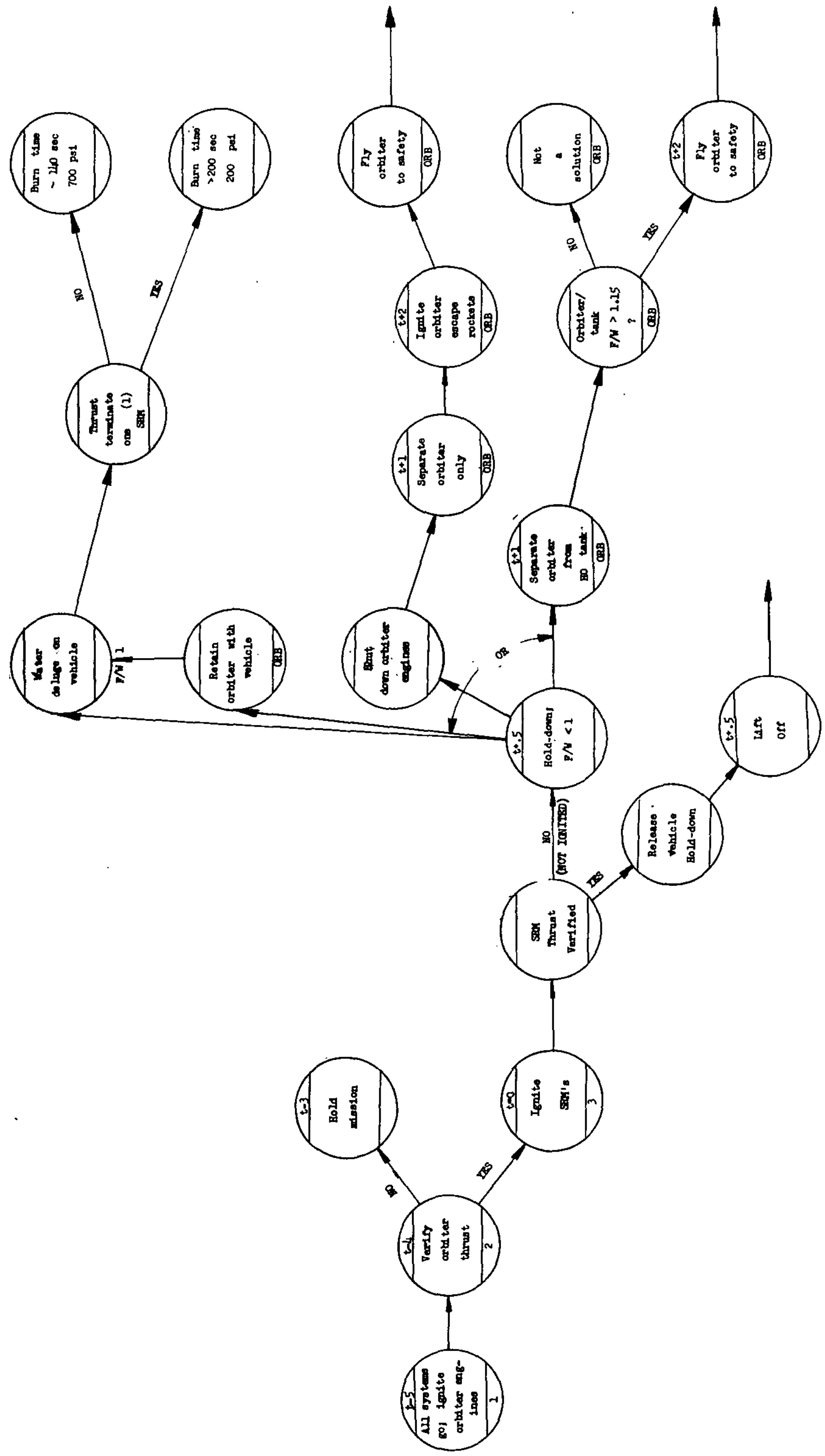

CRITICAL ABORT CONDITION
t ~ 30 SEC
V ~ 500 FT/SEC
ALT ~ 10,000 FT
t > 0.5 SEC
CRITICAL REGIME
ORBITER NOT MANEUVERABLE
ORBITER ESCAPE SYSTEM REQUIRED
ORBITER CAPABLE OF MANEUVERING TO SAFE LANDING



Thrust termination system design and performance are discussed in subsection 3.6. Considerations of impingement of the plume from the thrust termination ports on the liquid tank and Orbiter are discussed in subsection 5.4.4.

PAD ABORT

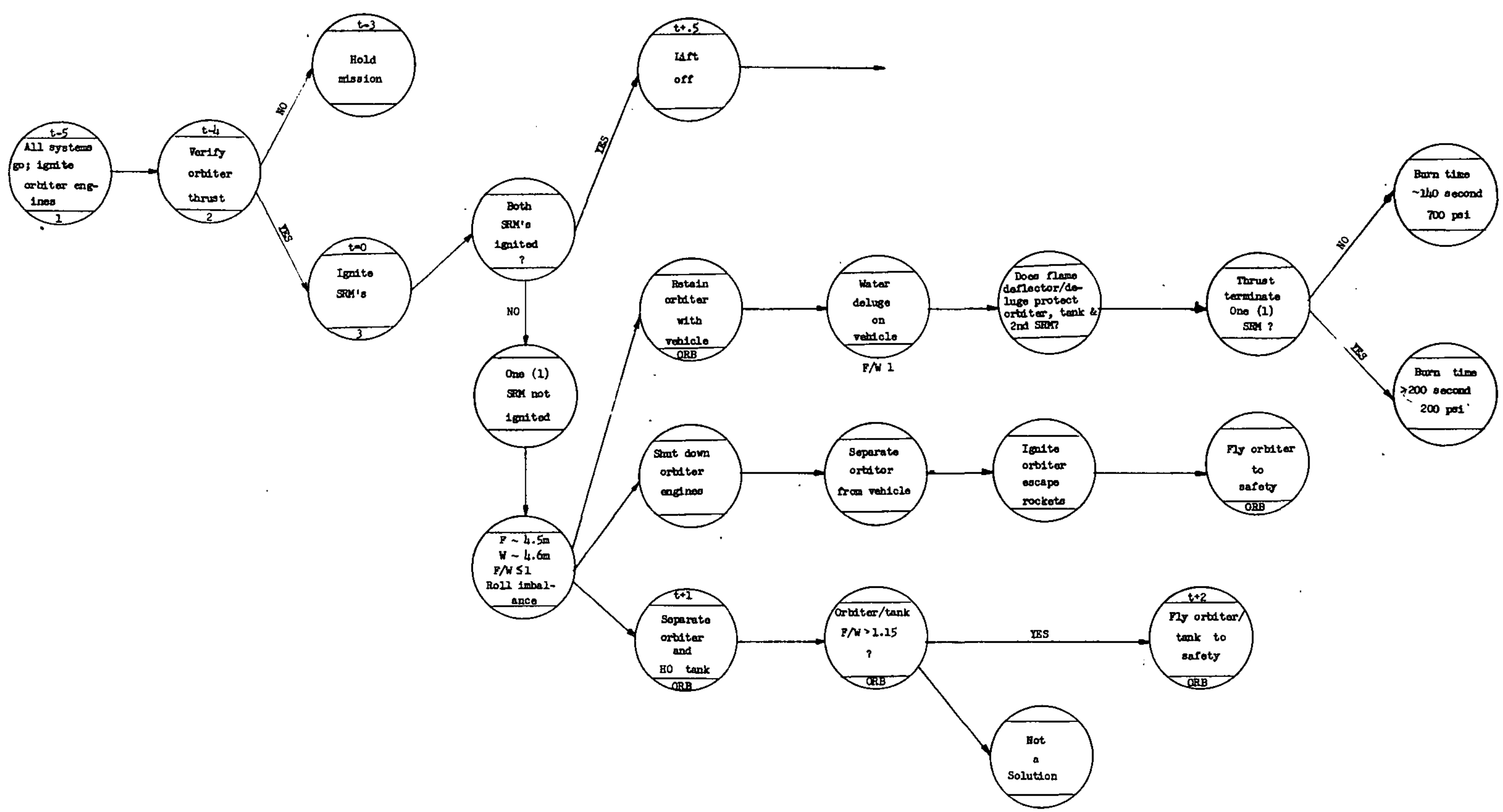
WITH HOLD - DOWN



-3 (4)

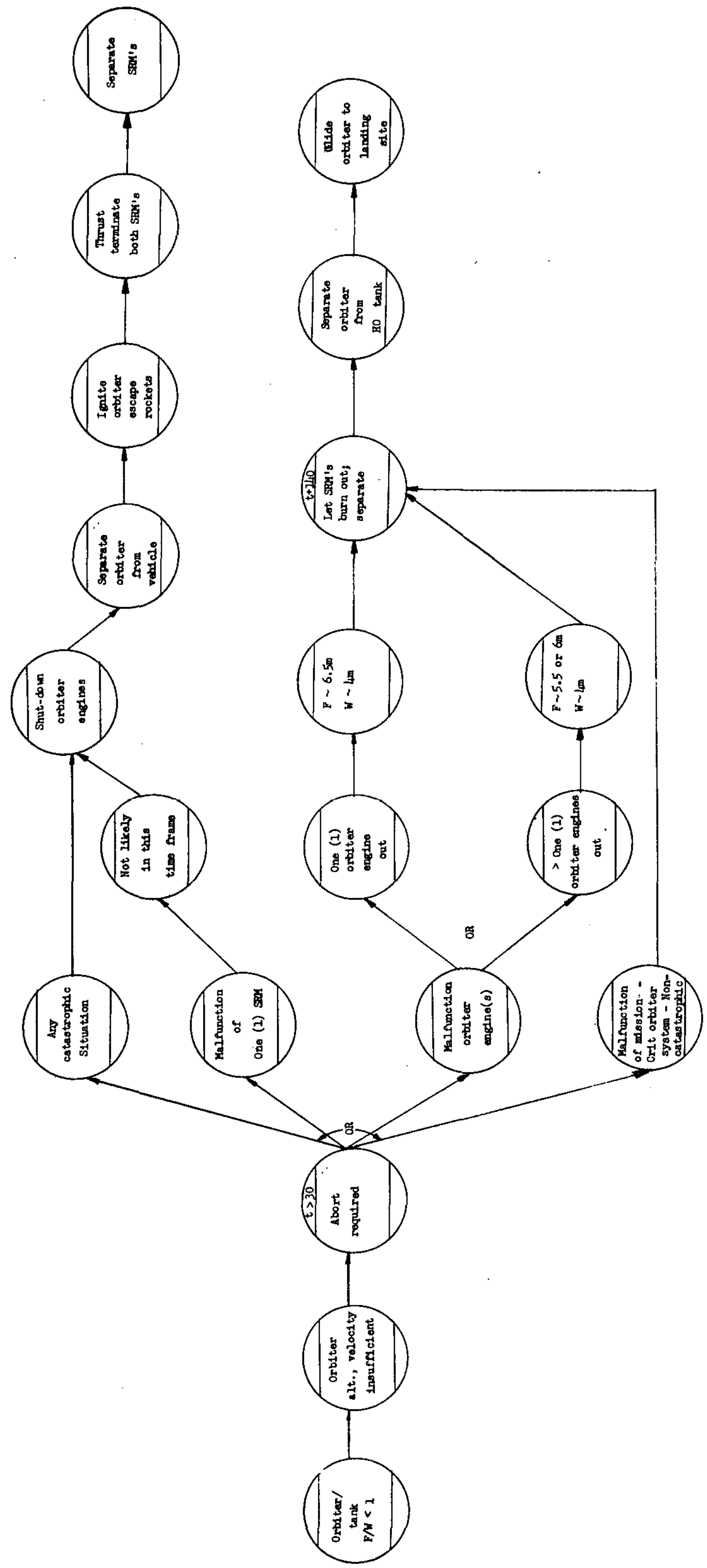
PAD ABORT

WITHOUT HOLD - DOWN



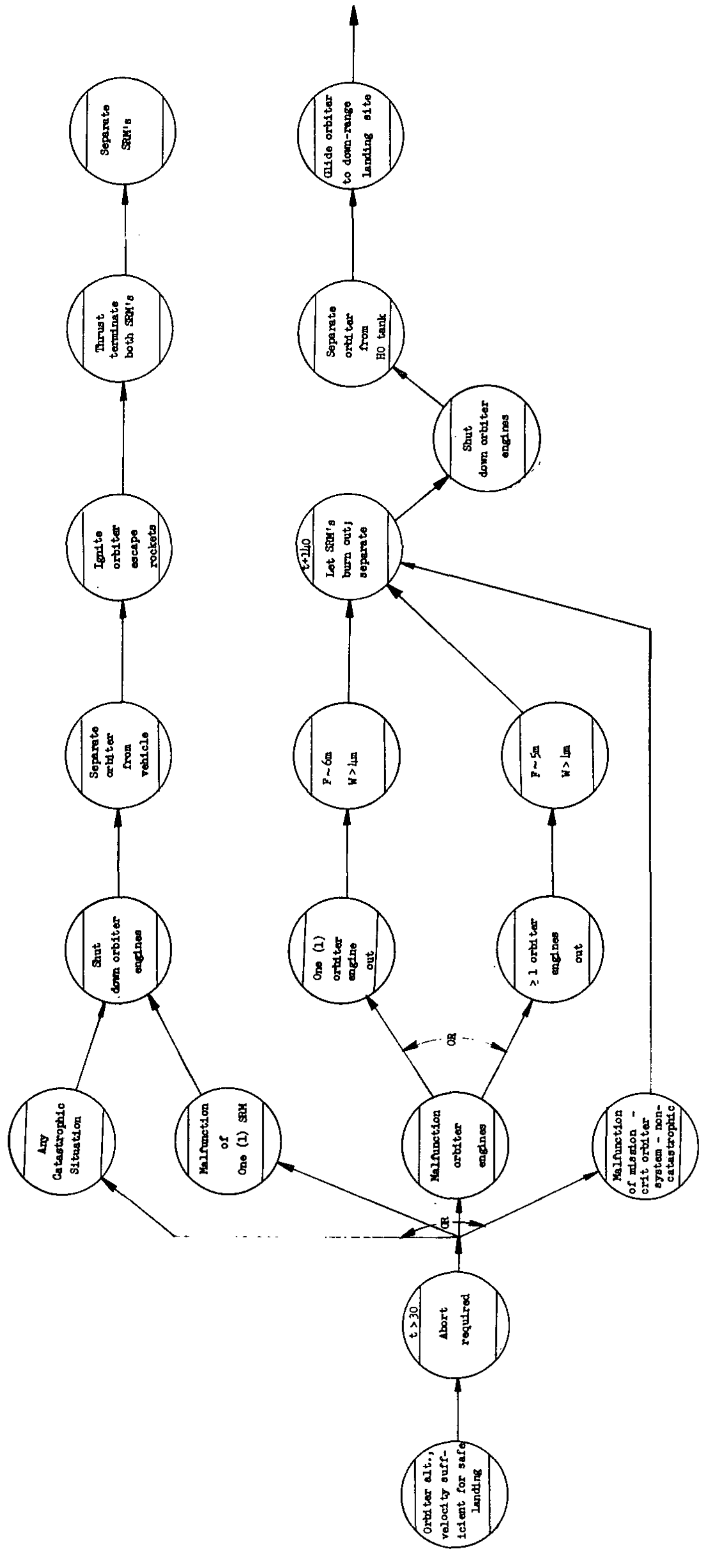
ABORT - EARLY FLIGHT

T~.5 TO T~30



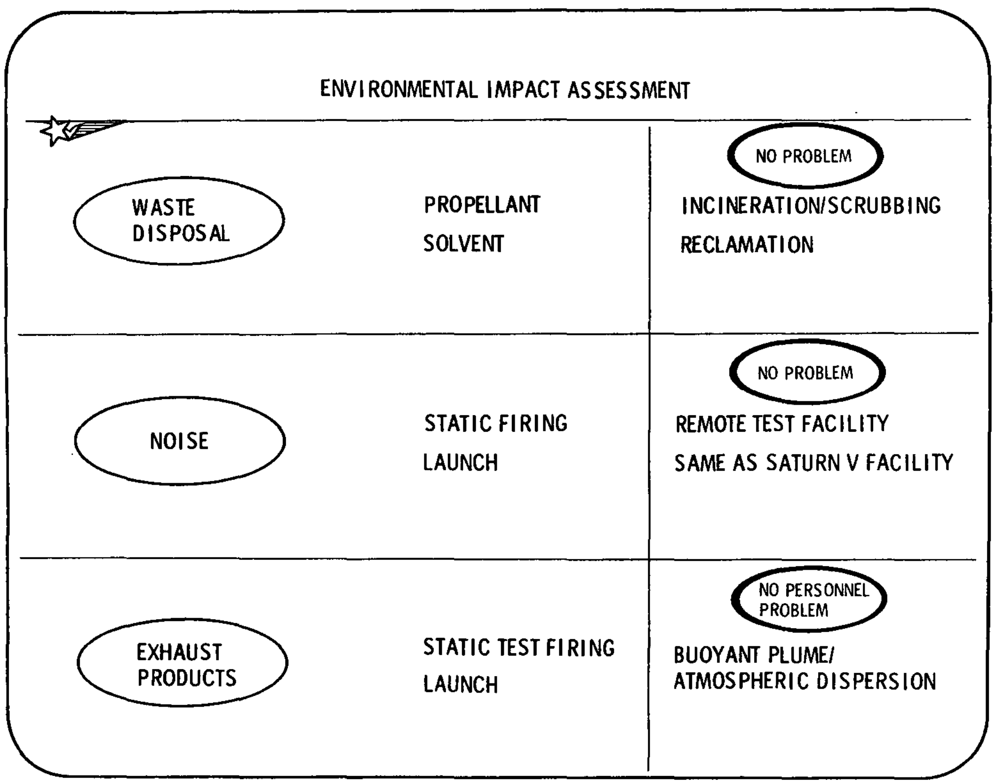
ABORT - LATER FLIGHT

T ~ 30 TO T ~ 140



5.1.3 Ecological Considerations

Lockheed Propulsion Company investigated the full scope of potential impact on the environment of the manufacture and launch of SRMs and assessed environmental issues. The launch plume is the area requiring further investigation concerning possible effects on plant life.

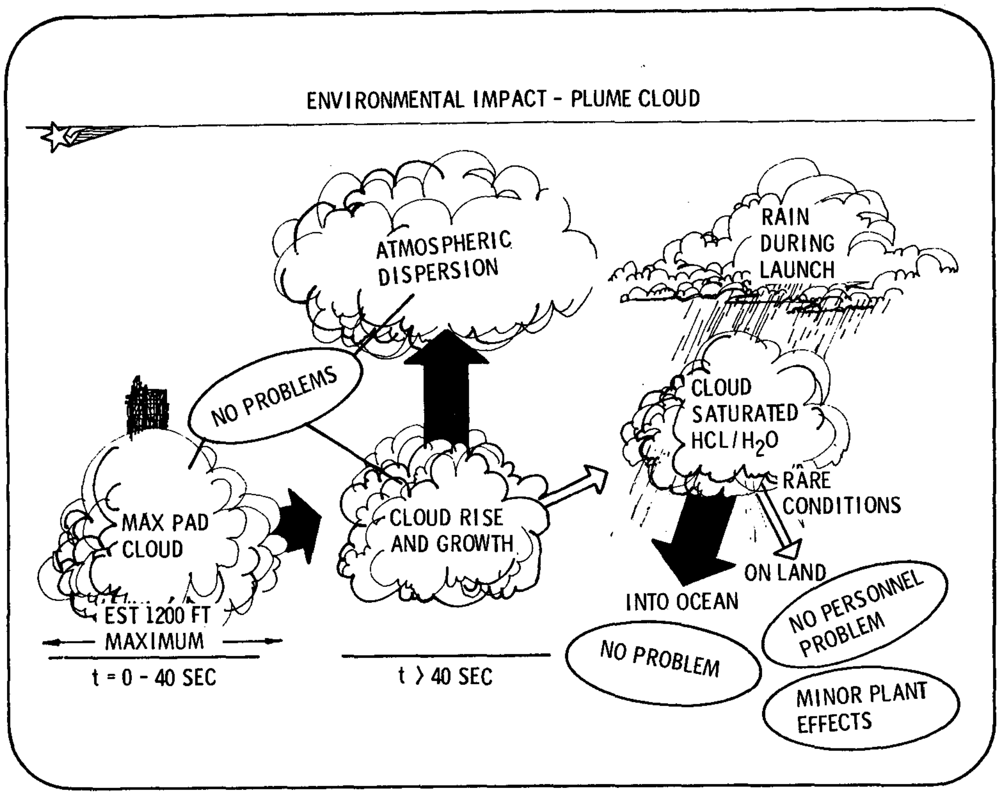


The specific chemical products of the launch plume have been identified and each individually assessed, as shown on the following chart. No basic problems exist. The possibility of the HCl returning to land and affecting plant life requires further analysis.

ENVIRONMENTAL IMPACT CONSIDERATIONS
LAUNCH EXHAUST PRODUCTS

SPECIE	CONCERN	CONCLUSION	SOURCE
Al_2O_3	PARTICULATE FALLOUT	NO PROBLEM LESS THAN TYPICAL URBAN	McDAC
H_2O, CO_2, CO	UPPER ATMOSPHERE	NO PROBLEM 35% OF SATURN V	NASA
N_2	NONE	NO PROBLEM MAJOR CONSTITUENT OF AIR	
HCl	TOXICITY	NO PERSONNEL PROBLEM PLUME RISE, ATMOSPHERIC DISPERSION	TITAN III DATA

As shown on the following chart, the launch plume sequence has been analyzed (and matched with existing experimental data). Two independent sets of conditions must occur for the fall-out to cause even minor plant effects. No personnel problems exist.



Conclusions. It was concluded that no personnel problems exist. The possibility of effects on plant life is rated as highly unlikely because it is necessary for both rain and on-shore wind conditions to be present during launch to cause exposure of plant life to hydrochloric acid from the exhaust plume. Additional details on LPC's environmental impact study are given in Appendix B.

5.2 SYSTEM SAFETY/HAZARDS ANALYSIS

The following pages present the results of a safety/hazard analysis.

HAZARD IDENTIFICATION AND LEVEL ANALYSIS

BOOSTER BUILDUP

DATE ______ PAGE 1 OF 8 REV. ______

ITEM NO.	HAZARD IDENTITY	HAZARD PRODUCING FACTOR	HAZARD EFFECT	SAFETY LIFE CYCLE MODE	HAZARD LEVEL	HISTORY OF OCCURRENCES
1.	Fire	A. Electrostatic, ferrous metal, electrical sparks B. Inadequate control of RF C. Lightning	Personnel injury/death. Damage to buildup facility	Booster build-	A	
2.	Faulty Booster	D. Improper assembly E. Buildup area fire F. Hoisting/handling equipment malfunction G. Personnel error	(D) Faulty/erratic Booster Aborted mission	(D) Launch flight	A	

PREVENTIVE MEASURES

1. Use propellants and components insensitive to electrostatic/RF environments.
2. Incorporate effective bonding/grounding between all booster elements.
3. Maintain strict grouding discipline during all booster assembly, test and handling operations.
4. "Human engineer" all equipment to reduce operator confusion/errors.
5. Carefully screen and train all assigned personnel.
6. Provide buildup facility with a quick reaction, massive innundating fire dowsing system.

HAZARD IDENTIFICATION AND LEVEL ANALYSIS

ACCIDENTAL BOOSTER INITIATION

DATE ______ PAGE 2 OF 8 REV. ______

ITEM NO.	HAZARD IDENTITY	HAZARD PRODUCING FACTOR	HAZARD EFFECT	SAFETY LIFE CYCLE MODE	HAZARD LEVEL	HISTORY OF OCCURRENCES
1.	Accidental Booster Initiation	A. Booster dropped/impacted during hoist/transport B. Test equipment malfunction/improper procedures. C. Personnel error	Personnel injury/death Damage to launch/transport/storage facility. Propulsive booster damage	Booster handling/transport/ordnance initiation/test	A	

PREVENTIVE MEASURES

1. Diligently test and maintain all components of hoisting/handling/transport equipment. "Human engineer" equipment controls to reduce operator confusion.
2. Carefully screen and train all assigned personnel.
3. Perform no voltage/stray voltage tests prior to installation of booster initiation items.

HAZARD IDENTIFICATION AND LEVEL ANALYSIS
BOOSTER HOISTING AND TRANSPORT HAZARDS

DATE ______
PAGE 3 OF 8
REV. ______

ITEM NO.	HAZARD IDENTITY	HAZARD PRODUCING FACTOR	HAZARD EFFECT	SAFETY LIFE CYCLE MODE	HAZARD LEVEL	HISTORY OF OCCURRENCES
1.	Booster damage	Hoist/transporter failure/malfunction	Personnel injury/death. Facility damage. Program delay	Booster movement	A	
2.	Booster damage (personnel error)	Hoist/transporter not properly operated. Personnel not properly trained/interested in assignment.	Personnel injury/death Facility damage. Program delay	Booster movement Booster movement	A	

PREVENTIVE MEASURES

1. Careful design, manufacture, assembly and test of all components of hoisting/transporter components.
2. Thorough screening and training of operating personnel.
3. "Human engineer" all equipment to reduce operation confusion/errors.

HAZARD IDENTIFICATION AND LEVEL ANALYSIS
TVC COLD GAS SYSTEM HAZARDS

DATE ______
PAGE 4 OF 8
REV. ______

ITEM NO.	HAZARD IDENTITY	HAZARD PRODUCING FACTOR	HAZARD EFFECT	SAFETY LIFE CYCLE MODE	HAZARD LEVEL	HISTORY OF OCCURRENCES
1.	Pressure tank rupture	Thermal/vibration environment	Loss of TVC/vehicle control.	Ground/launch/flight operation	A	
2.	Flex-hose/high pressure fitting failure	Faulty material/construction	Damage to booster. Aborted mission			
3.	Inadvertant activation of system	Personnel error	Personnel injury. Delay of mission	Installing or testing system	A	

PREVENTIVE MEASURES

1. Careful design, manufacture, assembly and proof test of the system.
2. Proof test system in actual environment of thrusting booster.
3. Provide system blocking components to eliminate possibility of activation because of human error.

HAZARD IDENTIFICATION AND LEVEL ANALYSIS

LAUNCH PHASE HAZARDS

DATE ____
PAGE 5 OF 8
REV. ____

ITEM NO.	HAZARD IDENTITY	HAZARD PRODUCING FACTOR	HAZARD EFFECT	SAFETY LIFE CYCLE MODE	HAZARD LEVEL	HISTORY OF OCCURRENCES
1.	Failure of 1 (or more booster(s) to ignite	Electrical/ordnance failure	Abort/thrust term/fire/ vehicle damage or loss/ facility damage. Lives of crew & passengers in great danger.	Attempted launch	A	
2.	Booster case/nozzle	Booster case/nozzle anomaly	Abort/thrust term/fire/ vehicle damage or loss/ facility damage. Lives of crew & passengers in great danger.	Attempted luanch		
3.	Inadvertant thrust term. /separation of booster(s)/orbiter	Malfunction of electrical/ ord. TT or booster separation systems	Abort/thrust term/fire/ vehicle damage or loss/ facility damage. Lives of crew & passengers in great danger.	Attempted launch		
4.	Accoustic noise/ ignition shock	Possible crew impairment/ . vehicle damage.	Subsequent failure or reduced reliability of crew/ vehicle.	launch		
5.	Cold gas TVC mal-function	Faulty inputs to TVC. Component failure/malfunction in TVC system.	Loss of directional control of vehicle. Thrust termination. Loss of vehicle. Lives of crew & passengers in great danger.			

PREVENTIVE MEASURES

1. Provide high reliability personnel escape system. Functioning capability zero altitudes to booster bornout altitude.
2. Provide redundant ignition thrust termination and separation systems (electrical and ordnance).
3. Conduct unmanned flights to evaluate affects of ignition shocks and acoustic noise on personnel and vehicle.

HAZARD IDENTIFICATION AND LEVEL ANALYSIS

BOOST PHASE HAZARDS

DATE ____
PAGE 6 OF 8
REV. ____

ITEM NO.	HAZARD IDENTITY	HAZARD PRODUCING FACTOR	HAZARD EFFECT	SAFETY LIFE CYCLE MODE	HAZARD LEVEL	HISTORY OF OCCURRENCES
1.	Booster case/nozzle failure	Booster case/nozzle anomaly	Thrust termination. Depending on time of occurrence, probable mission abort. Probable great danger for orbiter personnel.	Boost phase	A	
2.	Inadvertant booster/ orbiter separation	Separation system malfunction	Thrust termination. Depending on time of occurrence. probable mission abort. Probable great danger for orbiter personnel.	Boost phase	A	
3.	Vehicle damage from nozzle thermal and/or acoustic outputs	Thermal/acoustic protection system failure	Thrust termination. Depending on time of occurrence, probable mission abort. Probable great danger for orbiter personnel.	Boost phase	A	
4.	Unrecoverable maneuver	Assymetric thrust TVC failure	Thrust termination. Depending on time of occurrence. Probable mission abort. Probable great danger for orbiter personnel.	Boost phase	A	

HAZARD IDENTIFICATION AND LEVEL ANALYSIS

BOOSTER SEPARATION HAZARDS

DATE ______
PAGE 7 OF 8
REV. ______

ITEM NO.	HAZARD IDENTITY	HAZARD PRODUCING FACTOR	HAZARD EFFECT	SAFETY LIFE CYCLE MODE	HAZARD LEVEL	HISTORY OF OCCURRENCES
1.	Booster separation failure (one or more boosters)	Booster separation failure	Aborted mission. Possible emergency separation of orbiter	Booster separation	A	
2.	Destructive/damage to tank/orbiter during booster separation	Booster separation failure	Aborted mission. Possible emergency separation of orbiter.	Booster separation		

HAZARD IDENTIFICATION AND LEVEL ANALYSIS

ABORT HAZARDS

DATE ______
PAGE 8 OF 8
REV. ______

ITEM NO.	HAZARD IDENTITY	HAZARD PRODUCING FACTOR	HAZARD EFFECT	SAFETY LIFE CYCLE MODE	HAZARD LEVEL	HISTORY OF OCCURRENCES
1.	Orbiter damage	Loads applied to orbiter as result of abort environment, i.e., unrecoverable manuevers, Booster separation anomalies-TT debris-Acoustic and thermal environment	Reduced reliability of orbiter.	Abort	A	

5.3 SYSTEM INTEGRATION

The data on system integration are presented in subsection 4.2.2 of this book.

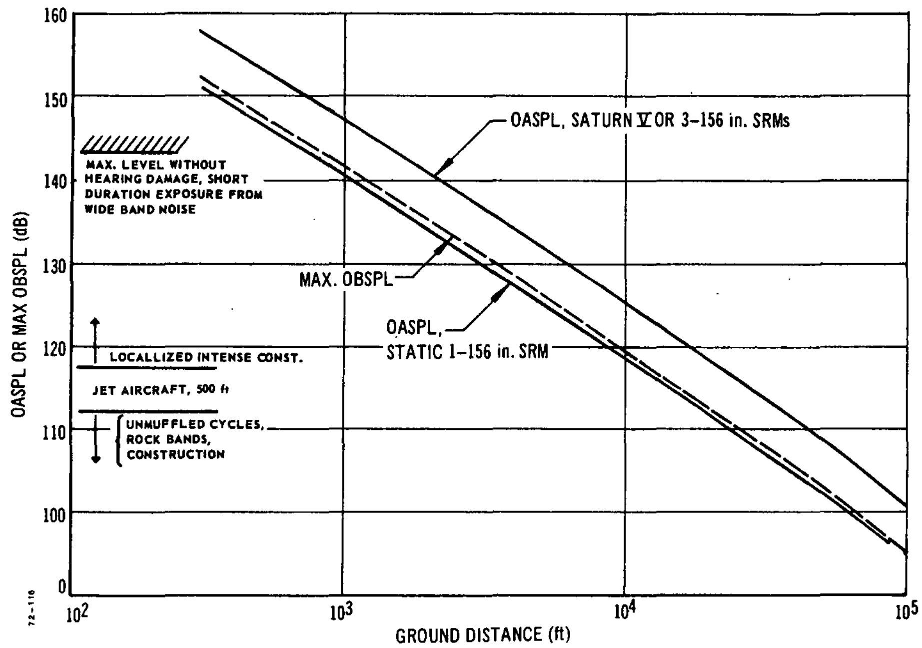
5.4 FLIGHT CHARACTERISTICS VERIFICATION

5.4.1 SRM Acoustic Field

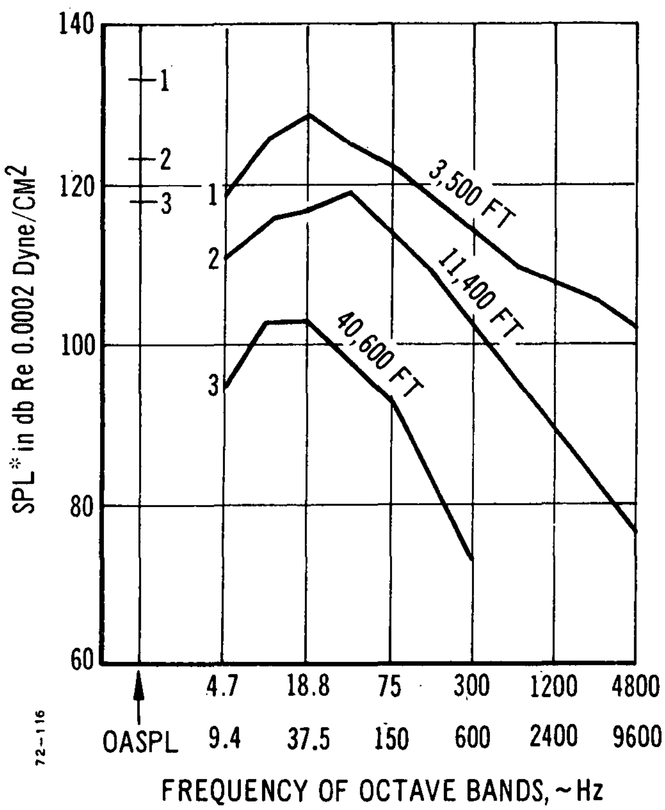
The major source of SRM noise appears to be the jet noise and its interaction with the ground and atmosphere. Both the acoustic power emitted and the frequency spectrum of the noise are affected by the size (thrust level) and the specific impulse of the rocket engine, as well as by design details. The nature of the noise may be described as intense, relatively short, composed predominantly of low frequencies, and infrequent.

The noise level for Space Shuttle launches using SRM boosters is expected to be equivalent to that of Saturn V launches. A summary of overall sound pressure level (OASPL) projections is shown on page 5-25. Both OASPL and maximum octave band sound pressure level (OBSPL) data are shown for Saturn V as a function of distance from the plume centerline. The expected OASPL for a static test of one 156-inch SRM booster is also shown and is estimated from previous static test data of motors of this size. Frequency spectra distributions of sound pressure level for Saturn V launches are shown on page 5-25.

Additional acoustic data are contained in Datacraft Report No. 219, "Acoustic Environment of Very Large Solid Propellant Motors," September 1967 (Contract No. NAS 8-11760). This report covers near- and far-field acoustic data acquisition and analysis of five static firings of large solid motors, two of which were 156-inch size (one of which was an LPC test) and three 260-inch size.



Far Field Sound Pressure Levels for Saturn V



Maximum Free-Flight, SPL Spectra for Saturn Test

5.4.2 Unsteady Aerodynamic Considerations for the Space Shuttle-SRM Designs

Unsteady aerodynamic effects have been shown to be capable of dominating the aeroelastic stability of space launch vehicles.[1,2] The unsteady aerodynamic loads are of two types: buffet loads, which are independent of body motion; and motion-dependent, separated-flow, loads. The former may result in panel flutter or may excite the free-free bending mode oscillation. The latter, the motion-dependent loads, can result in aerodynamic undamping of the lower elastic modes and possibly ultimate failure of the vehicle. Therefore, it is prudent design practice to avoid the unsteady flow phenomena that can cause adverse motion-dependent loads. Presently, five potentially dangerous flow phenomena have been identified on the Space Shuttle-SRM booster designs. Three of these may be avoided by following a few simple guidelines. The other two are inherent in the basic design and cannot be eliminated. Their seriousness can be assessed, and minimized, by a combination of wind tunnel tests and analysis during development. A discussion of these five unsteady flow phenomena follows:

(1) Collapsing Separation

At high subsonic speeds, the terminal normal shock that resides just aft of the shoulder of slender cone-cylinder geometries can be a source of sudden discontinuous change in the shoulder load.[3,4] The leeward side boundary layer over the cone-cylinder weakens with increasing angle of attack until it can no longer support the terminal normal shock. The shock then jumps to the cone-cylinder shoulder, causing gross leeside separation and a discontinuous change in the shoulder load. The effect in the shoulder load will result in aerodynamic undamping of the elastic mode to such a degree that a limit cycle oscillation will result.[5] The magnitude of the limit cycle is determined by the balance between the structural damping and the aerodynamic undamping. However, the collapsing separation can be easily avoided. A biconic configuration[5] shown on the following page can delay the collapse to an

(1) Woods, P., and Ericsson, L. E., "Aeroelastic Considerations in a Slender, Blunt-Nose, Multi-stage Rocket," Aerospace Engineering, May 1962

(2) Ericsson, L. E., and Reding, J. P., "Analysis of Flow Separation Effects on the Dynamics of a Large Space Booster," J. Spacecraft and Rockets, Vol 2, No. 4, 1965, pp 481-490

(3) Robertson, J. E., and Chevalier, H. L., Characteristics of Steady-State Pressures on the Cylindrical Portion of Cone-Cylinder Bodies at Transonic Speeds," AEDC TDR 63-104, August 1963

(4) Chevalier, H. L., and Robertson, J. E., Pressure Fluctuations Resulting From Alternating Flow Separation and Attachment at Transonic Speeds, AEDC TDR 63-204, November 1963

(5) Ericsson, L. E., and French, N. J., The Aeroelastic Characteristics of the Saturn IB SA-203 Launch Vehicle, LMSC M-37-66-2, April 1966

angle of greater than 16 degrees. A small region of separation occurs at the juncture of the 25 and $12\frac{1}{2}$-degree frustums. The low-energy boundary layer flow is trapped in the recirculation region. The result is a stronger boundary layer downstream that is better able to support the coupled effects of the terminal normal shock and the adverse pressure gradient due to the shoulder expansion.

(2) Boattail Separation

A phenomenon similar to collapsing separation can also occur on boattails (rearward sloping surfaces) at transonic speeds. The terminal normal shock that occurs at the downstream end of the boattail will move forward on the leeward side at angle of attack. The result is a negative, attitude-sensitive load (page 5-28) that can cause aerodynamic undamping of the lower elastic modes. This is typically a slender boattail effect. If the separation is fixed by a steep rearward slope, the negative load due to the disparity between windward and leeward side shock positions is avoided (page 5-28). It is believed that the 3-degree boattail present in the Series-Burn Space Shuttle-SRM configuration will avoid this adverse effect, because it is largely compensated for by boundary layer growth.

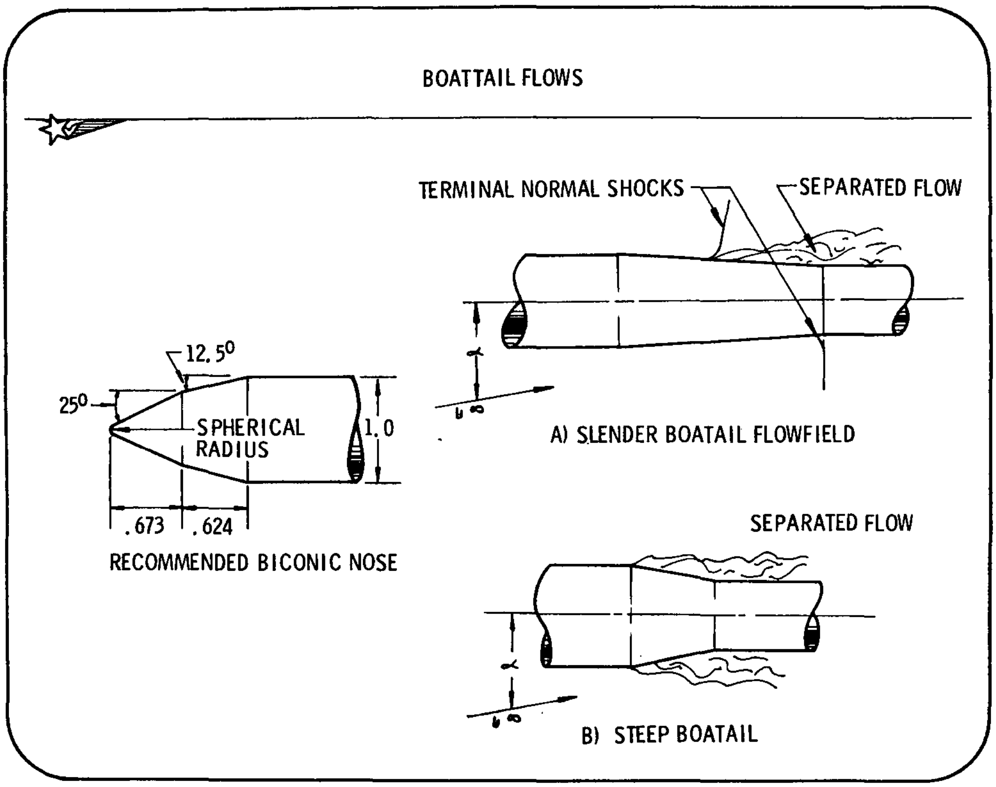
(3) Flare-Induced Separation

Large moments (as large as 200,000 ft/lb) result from aerodynamic loads in the SRM nozzles. Thus, it is necessary to shield the nozzles with a tail flare. Of course, the adverse pressure gradient caused by the flare shock can cause flow separation at transonic speeds and contribute to aerodynamic undamping. Experience with the Apollo-Saturn vehicles has shown that the flare-induced loads are not necessarily serious because the negative induced load in the region of the separation shocks tends to cancel the effect of the increased flare lift. However, it seems prudent to try to avoid the separation if possible. This can easily be accomplished by keeping the flare angle less than about 15 degrees (in the present design the tail flare angle is 10 degrees).

(4) Shock Impingement

The shock impingement between parallel stages is another possible source of aerodynamic undamping.[6] This includes both the interference between the shuttle and the SRMs in the parallel-burn

(6) Reding, J. P., and Ericsson, L. E., "Unsteady Aerodynamic of Manned Space Vehicles Past, Present, and Future," Transactions of the First Western Space Congress, Santa Maria, Cal, October 27-29, 1970, pp 882-893


BOATTAIL FLOWS
TERMINAL NORMAL SHOCKS
SEPARATED FLOW
A) SLENDER BOATAIL FLOWFIELD
12.5°
25°
SPHERICAL RADIUS
1.0
.673
.624
RECOMMENDED BICONIC NOSE
SEPARATED FLOW
B) STEEP BOATAIL

configuration and between the Shuttle Orbiter and external propellant tank. However, shock impingement effects are common to virtually all shuttle configurations and cannot be avoided. Quasi-steady techniques may be used to evaluate the seriousness of any adverse interference effects and to suggest fixes to alleviate any problems.

(5) Orbiter Wake Interference

The wake from the Orbiter attaching on the SRMs in the series-stage configuration presents yet another possible detrimental unsteady aerodynamic interference effect that cannot easily be eliminated. The attaching wake provides a time-dependent, attitude-sensitive load at reattachment and can possibly affect the Orbiter loads through the wake recirculation. Again the problem cannot be assessed without a knowledge of the structural characteristics (mode shape) and the experimentally determined aerodynamic loads.

Conclusions. Of the five possibly detrimental, unsteady, motion-dependent, aerodynamic-flow phenomena identified on the Space Shuttle-SRM configurations, three have been avoided in the present designs. The other two, namely shock impingement and orbiter wake attachment, cannot be avoided. However, the dynamic effects of separated flow can be analyzed using static experimental data. Thus, the adverse interference effect can be minimized or eliminated during the vehicle development.

5.4.3 Thermal Effects

5.4.3.1 156-Inch Parallel-Burn SRM

The two-motor, parallel-burn booster configuration introduces thermal problems in two areas, as follows:

(1) Tank Base - The liquid hydrogen tank base is exposed to heat radiation (18 Btu/ft^2-sec) from the plumes of the booster exhaust in addition to that from the orbiter exhaust. The spatial arrangement of all nozzles indicates that convective heating will be minimal. The aft dome of the tank will require approximately 1500 pounds of insulation (0.05-inch-thick cork or equivalent).

(2) Tank and 156-Inch SRM Forward Interface - The interaction of the bow shock from the SRM on the tank will impose insulation requirements on the tank similar to the requirements imposed by the orbiter shock.

5.4.3.2 156-Inch Series-Burn SRM

The series configuration imposes no thermal conditions on the orbiter tanks.

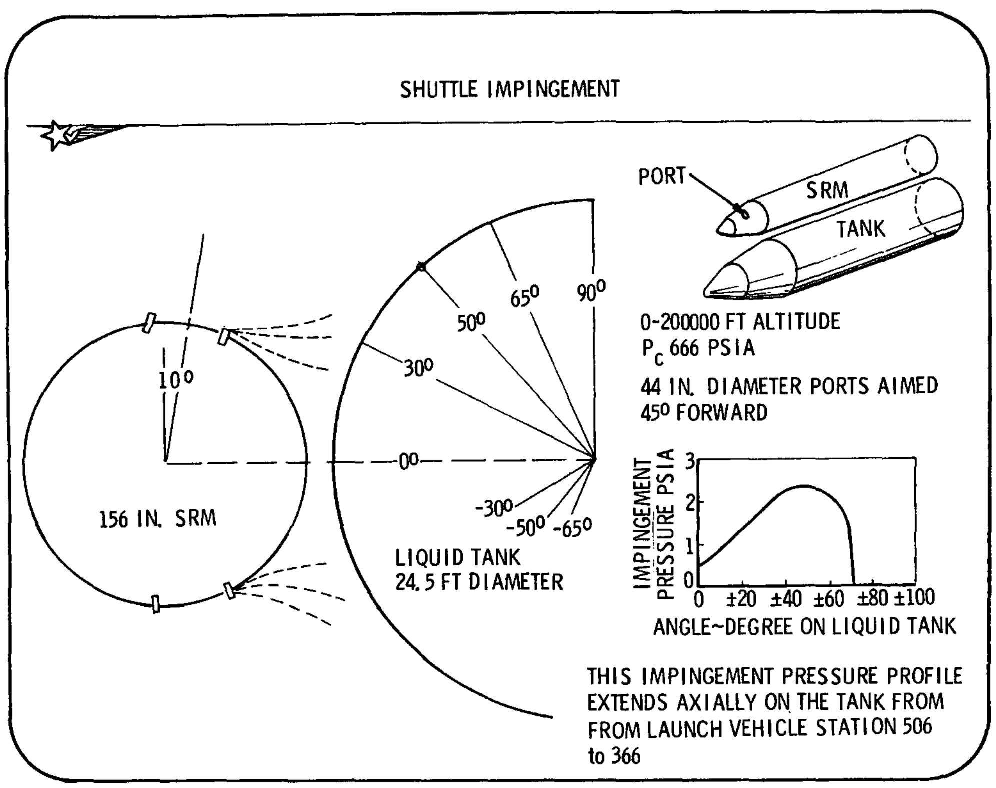
The cluster of three boosters will cause a base recirculation and heating problem for the motor bases and control systems. A skirt extension near the nozzle exit with base plates closing off much of the open areas around the nozzle exits will minimize base recirculation. Exclusion of the skirt will result in substantial base heating levels with secondary combustion occurring. Assessment of insulation requirements will be required for the entire base assembly and control components.

Reverse flow into the base region for a single nozzle design is driven by the free stream/exhaust flow interaction. As the slip stream momentum increases, the flow reversal increases to a maximum, then decays as the slip stream momentum goes to zero. This is generally accompanied by base combustion due to the hydrogen-rich exhaust entrainment with the oxygen in the slip stream. The net effect on Poseidon-type configurations yields heating rates of 25 Btu/ft^2-sec and gas temperatures of 3000^{0}F.

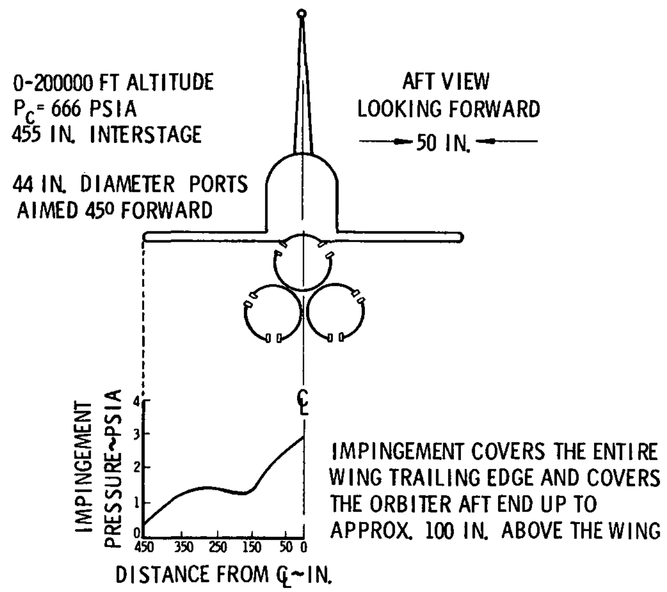
Multiple nozzle flow reversal is driven by the interaction of adjacent exhaust flow and low pressures induced on the base by the free stream. This type of base flow results in much higher heating than occurs with a single nozzle. In general, the heating rates experienced on Polaris-type configurations ranged from 50 to 150 Btu/ft^2-sec. These rates are very geometry-dependent, however, and considerable work needs to be done before predictions can be made on a tri-motor cluster.

5.4.4 Shuttle Impingement -- Thrust Termination

Thrust termination plume impingement pressures on the liquid tank and orbiter have been estimated for two 156-inch SRM booster configurations. The following figures show the pressures and areas covered by impingement. Impingement pressure was estimated by means of the Modified Lee's Newtonian Method and a sonic port thrust termination plume as established by a Method of Characteristics.

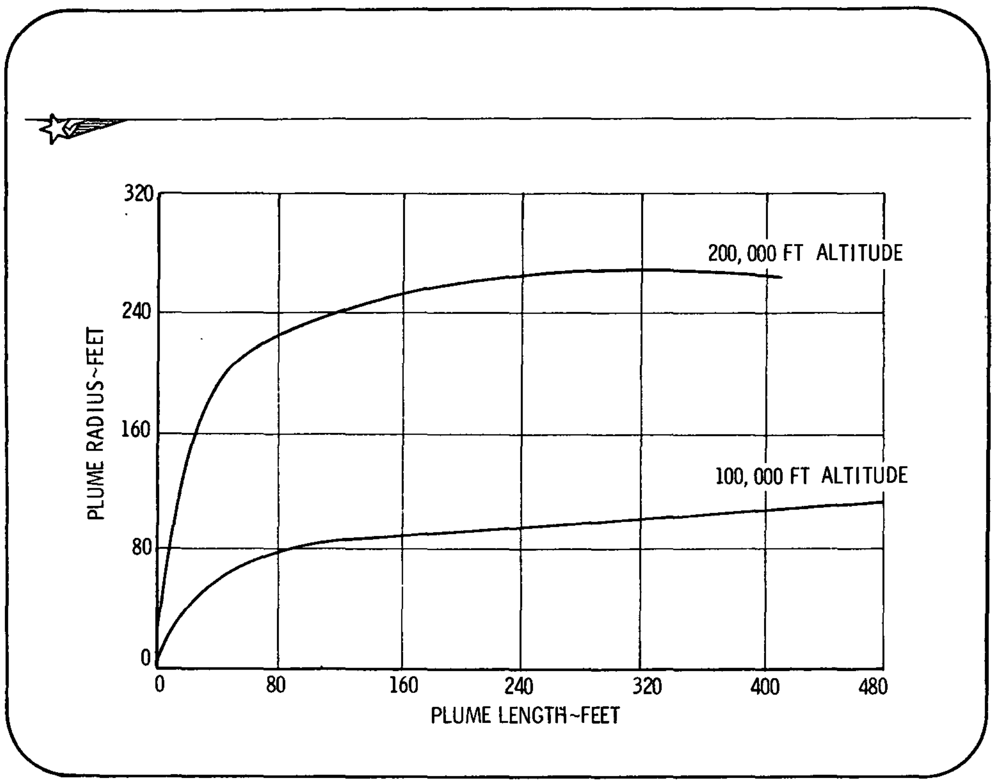


SHUTTLE IMPINGEMENT
ABORT THRUST TERMINATION



Thrust termination port plug and debris. The possibilities that the termination port plugs may impact the orbiter in the series-burn, three-SRM configuration or that debris may impact the liquids tank in the parallel-burn, two-SRM configuration were considered. A study of this potential problem resulted in the following conclusions:

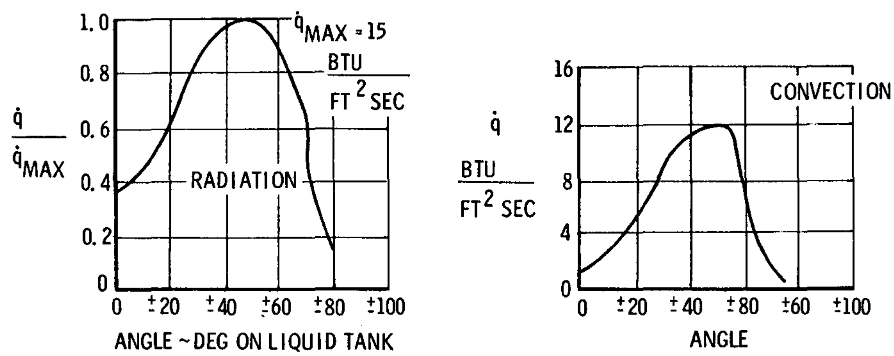
- Limited experience on Poseidon first-stage ignition shows that the nozzle plug follows the plume centerline.
- Analysis of Poseidon thrust termination plug trajectory indicates that the plug reaches a velocity of 1000 to 2000 ft/sec within a distance of 5 to 10 feet from the port.
- Plug debris for the SRMs is expected to stay within the bounds of the termination plume as shown below.



The graphs below show the heating effects due to plume impingement from the termination ports.

IMPINGEMENT HEATING

"PARALLEL" SRM CONFIGURATION



"SERIES" SRM CONFIGURATION

HEATING OF ORBITER WINGS TRAILING EDGE

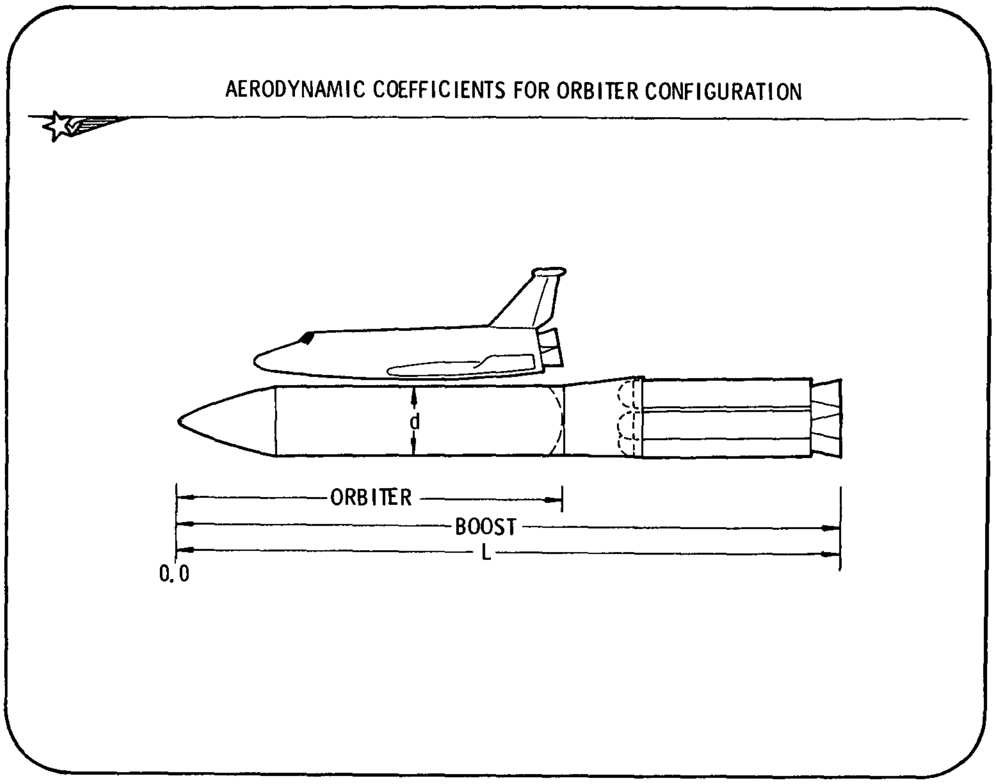
$\dot{q}_{CONVECTION} \approx 20$ BTU/FT^2 SEC

$\dot{q}_{PARTICLE\ IMPINGEMENT} \approx 300$ BTU/FT^2 SEC (Al_2O_3 IMPINGEMENT)

NOTE:

THE ORBITER IS INSULATED FOR REENTRY SO TERMINATION PLUME HEATING MAY BE WITHIN DESIGN LIMITS

5.4.5 Aerodynamic Coefficients for Orbiter Configuration



Stability and drag characteristics were determined for both the boost and orbiter configurations at a Mach number of 1.4 and a corresponding dynamic pressure of 630 psf. Report DMS-DR-1183, Aerodynamic Stability and Control Characteristics, relating to a 0.0036-scale Boeing RS-1C/MSC-040A orbiter at Mach numbers of 0.6 to 5.0, was used as a source for the coefficients. The complexity of the configuration and the preliminary nature of the analysis require that the calculated characteristics be used with restraint:

	Orbiter	Boost	
C_{N_x}	0.223/deg	0.283/deg	SREF = $\pi/4\ (d)^2$
XCP	33.5% L	34.2% L	XCP = %L from station 0.0
$C_{A_{forebody}}$	1.45	1.54	

Section 6

DESIGN ANALYSIS

6.1 STRUCTURE

Design of structural components during the study was based on review and modification of detailed designs prepared previously by LPC on 156-inch-diameter, large solid motor programs. The external loads presented in Appendix A will form the basis for detailed analysis during future phases of effort.

The primary structural design effort was conducted on the parallel-burn, heavy-payload configuration consisting of seven interchangeable segments, a forward closure containing two thrust termination ports and an igniter boss, and an aft closure with a symmetrical nozzle boss.

The motor case material is D6AC steel, heat-treated to an ultimate tensile strength of 225,000 psi. The nominal wall thickness of 0.460 inch is based on the allowable strength level and a 13-percent biaxial improvement as a result of the roll-forming technique used in segment fabrication. The 0.460-inch thickness provides a 1.40 safety factor above MEOP of 1000 psig. The case wall determination is presented in the Table on the following page.

The clevis joint used to join the segments is a combination of the tapered pin with face seal clevis joint used successfully on the LPC 120- and 156-inch large solid motor programs and the 120-inch Titan IIIC straight pin with band seal clevis joint. The tapered pin used in the LPC clevis joint is combined with the cylindrical seal used with the Titan IIIC clevis joint. The cylindrical seal was selected on the basis of greater experience with this type of seal. The tapered pin was selected over the straight pin in order to provide a wedging action at assembly, which will provide a tight joint independent of feature tolerances.

The tapered pin clevis joint design is basically the LPC 156-inch LSM design adjusted for differences in the material type and strength and the overall outside diameter of the 156-inch-diameter cylindrical section.

The nozzle attachment is a symmetrical bolted joint. The 10-degree angle of the nozzle is achieved through the nozzle attachment flange. Fasteners having a strength of 220,000 psi are used to attach the nozzle and igniter.

The igniter and thrust termination port cover attachments are bolted, reinforced bosses in the forward closure. The covers for the thrust termination ports and the igniter assembly are attached to the case with fasteners having a strength of 220,000 psi.

MOTOR CASE WALL THICKNESS DETERMINATION

	Parameter	Value	Source
1.	Pressure, nominal maximum at 80^0F (psi)	938	Predicted P-t curve
2.	Pressure, nominal maximum at 90^0F (psi)	952	$(\pi_p)_k$ 0.148%
3.	Pressure, variation at 90^0F (psi)	48	(3σ) - 5.0%
4.	Pressure, limit, MEOP (psi)	1000	952 + 48
5.	Pressure, proof (psi)	1100	(1.1) 1000
6.	Safety factor	1.40	
7.	Burst pressure	1400	(1.40) (1000)
8.	Pressure, design yield minimum (psi)	1275	$\frac{205,000}{225,000} \times 1400$
9.	Stress, minimum ultimate (psi)	225,000	D6AC properties
10.	Stress, minimum yield (0.2)(psi)	205,000	D6AC properties
11.	Biaxial gain (%)	13	Minuteman experience
12.	Biaxial ultimate strength (psi)	254,000	(1.13) (225,000)
13.	Biaxial yield strength (psi)	232,000	(1.13) (205,000)
14.	Temperature, maximum wall (^0F)	200	Internal heating, estimated
15.	Strength loss due to Item 14 (%)	2.0	MIL-HDBK-5
16.	Thickness, case wall, minimum (in.)	0.440	$\frac{(1400)\ (78)}{(254,000)\ (0.98)}$
17.	Thickness, case, nominal (in.)	0.460 ± 0.020	
18.	Ratio $\frac{\text{yield pressure}}{\text{proof pressure}}$	1.16	$\frac{1275}{1100}$

Maximum motor case stress at MEOP

$$S_m = \frac{PR}{t} = \frac{(1000)\ (78)}{0.440} = 177,000 \text{ psi}$$

Safety factor between MEOP and yield strength at maximum temperature

$$S.F. = \frac{(232,000)\ (0.98)}{177,000} = 1.28$$

6.2 MANUFACTURING, MATERIALS, AND PROCESSES

This subject is covered in Section 3.2 of Volume III, Program Acquisition Planning.

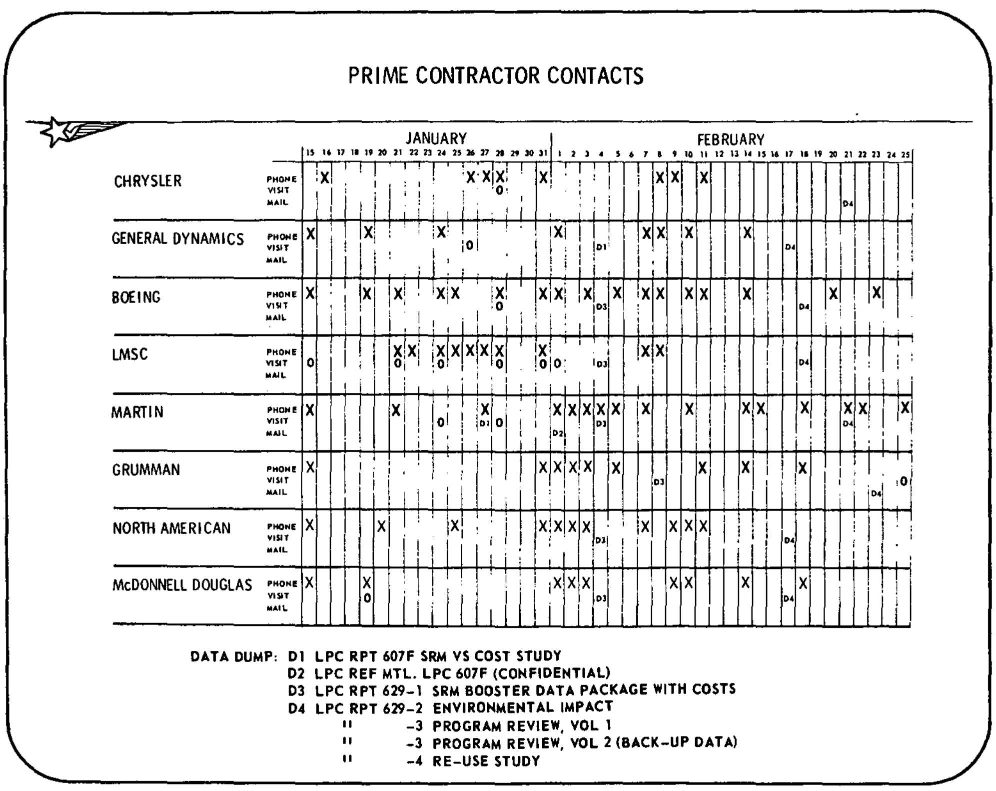
Section 7

PHASE B CONTRACTOR TECHNICAL SUPPORT

In compliance with Section 3.0 of Exhibit A, "Statement of Work", Lockheed Propulsion Company provided a full-time liaison staff to respond to data requests from Phase B system contractors. This staff visited the Phase B contractors to review data and provide technical support. These visits, and related telephone discussions, are shown in the following figure.

The LPC baseline was established after review of the design requirements received from each of the Phase B contractors. A summary of these requirements is presented in the following table. Specific designs presented by each contractor were then examined for conformance with the LPC baseline. These designs were reviewed with the contractors and mutually acceptable design solutions were evolved.

The preliminary data were delivered to each contractor on 4 February (7 February to Grumman Aerospace). The final data submittal was presented at each contractor's facility by the teams on 17-18 February.



DATA COMPARISON - SPACE SHUTTLE, PARALLEL BURN

Data Item	LMSC	MDAC	M-D	GDA	TBC	Chrysler	NASA	LPC
Payload weight (lb)	65K	65K	65K	65K (40K)	65K	65K		65K
Orbit	East	East	East	East (Polar)	East	East		East
GLOW (M lbs)	4.902	3.923		4.765	4.590			4.635
BLOW (M lbs)	3.044	2.142	2.700	2.899	2.788			2.835
OLOW (M lbs)	1.858	1.781	1.600	1.866	1.807	1.825		1.800
Maximum g	3	3						
Maximum Q	650	650	650	650	650	650		
Staging velocity (ft/sec)	5800	4211	6000	4747	5300	5000		
Staging altitude (ft)	180,000		170,000		140,000	127,000		
Orbiter thrust (vac) (lbf)	Not specified	1.410×10^6	1.410×10^6	1.410×10^6	1.4073×10^6			
F/W at liftoff	1.25	1.405	1.40	1.427		1.40		---
SRMs per vehicle	2	2	2	2	2	2		2
Staging rockets	Not specified		Fore and aft			Being considered		---

DATA COMPARISON - SRM

Data Item	LMSC	MDAC	M-D	GDA	TBC	Chrysler	NASA	LPC
Motor length (in.)	Not specified				1411			1494
Motor weight (lb)	1.497 x 10^6			1.176 x 10^6	1.394 x 10^6			1.385 x 10^6
Segment length (in.)	Not specified							160
Number of segments	Not specified			4	8 +2 domes			7 +2 domes
Propellant type	PBAN				HTPB			PBAN
Propellant weight (lb)	1.347 x 10^6	0.953 x 10^6	1.200 x 10^6	1.274 x 10^6	1.246 x 10^6	1.204 x 10^6		1.231 x 10^6
Burn time (sec)	138	125		123		134.8		138
I_T vac/motor (lbf-sec)	3.583 x 10^8	2.516 x 10^8	3.168 x 10^8	3.428 x 10^8	3.380 x 10^8	3.251 x 10^8		3.260 x 10^8
I_{sp}, vac (lbf-sec/lbm)	266.0	264	264	269	271.3	270		264.8
Mass fraction	0.90(SRM) 0.885 (stage)	0.890		0.9088	0.894			0.889[(a)] 0.901[(b)] 0.906[(c)]
F_{vac} initial (lbf)	2.749 x 10^6	2.26 x 10^6	2.77 x 10^6	3.25 x 10^6	2.69 x 10^6			2.942 x 10^6
Expansion ratio	9		5	10	10	9.8		8.3

(a) With TVC and TT (b) Without TVC (c) Without TVC and TT

DATA COMPARISON - SRM

Data Item	LMSC	MDAC	M-D	GDA	TBC	Chrysler	NASA	LPC
Case material	D6AC	D6AC or 200 grade		200 grade[a]	D6AC			D6AC
Fabrication method	Not specified							Roll form
UTS (Ksi)	200	200		225[a]				225
Safety factor (case)	1.4	1.25		1.4	1.28			1.4
Cant angle (deg)	11	15	< 10		0	10		10
TVC type	Gimbal		Gimbal		Flex. bearing mov. noz.			Lockseal gimbal
Actuation	N_2 blowdown							N_2 blowdown
Deflection (deg)	10	0	5		6	6		10
Slew rate (deg/sec)	15				5	20		15
Thrust termination	Yes	Yes	Yes	Yes	Yes	Yes (abort)		Yes
Recoverable	No					No		No
Destruct requirement			Yes		No	Yes		
Ignition				Aft end	Head end			Head end
Malfunction detection				Yes	Yes		Yes	

(a) Based on data for series burn

Section 8

RELATED EXPERIENCE

Since its founding in 1952, LPC has been identified with the progress and advancement of solid propellant rocket motor technology and has introduced many innovations now accepted as standards in the solid propellant industry. Among other achievements, LPC designed, built, and test-fired the world's first 120-inch-diameter solid rocket motor in 1962 and the first 156-inch-diameter solid rocket motors in 1964.

The 120-inch Applied Research Motor, developed for the Air Force under Contract 04(611)-8013, incorporated the first large-diameter, hot-formed elliptical dome, the first large-scale use of a segmented motor case, the first large-scale use of N_2O_4 in a secondary fluid injection TVC system, and set a new thrust record of 350,000 lbf.

Two years later, LPC designed and tested two 156-inch segmented solid rocket motors with jet tab TVC under Air Force Contract 04(695)-364. This program featured the first use of 18-percent nickel maraging steel for a large solid rocket motor case, the successful use of ablative plastic in nozzle throat, and the first refurbishment and reuse of a large-scale motor case. The first motor was fired in May 1964 and the second in September of the same year. At the time of the firings, these were the world's most powerful solid rocket motors.

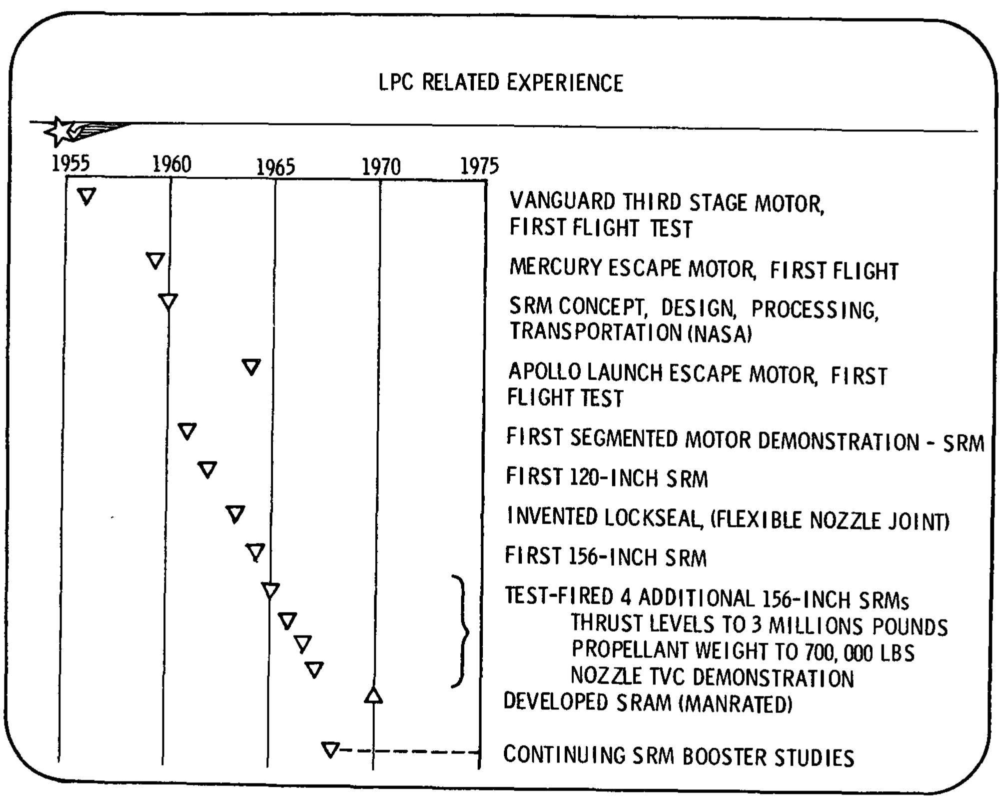
Two flightweight versions of the 156-inch large solid rocket motor were built and fired under Contract 04(695)-772 in a continuation of the Air Force program for development and demonstration of large SRMs and related technology. Both motors incorporated a nozzle-mounted liquid injection TVC system, a deep-submerged nozzle, a high-burn-rate propellant, and reused case components. The first, a 3,000,000-lbf thrust class segmented motor, was successfully fired in December 1965 and the second, a 1,000,000-lbf thrust class monolithic motor, in January 1966. The Terminal Contractor Evaluation Report released by the Air Force at the conclusion of this program stated in part:

> "The results obtained by the Lockheed Propulsion Company in satisfying the technical and schedule requirements of this contract are considered excellent in all respects. Two 156-inch-diameter motors of advanced configuration were designed, fabricated, and successfully tested well within the time period specified.
>
> "As a direct result of the contractor's efforts, several highly significant advancements in the large solid motor state-of-the-art were achieved. For example, large submerged nozzles were tested for the first time, thus permitting a significant reduction in overall

motor length. A nozzle-mounted omniaxial thrust vector control system was developed and tested at a motor thrust level twice that of any previous system. The high burn rate propellant (0.8 in./sec) developed is a particularly significant advancement in that the increased burn rate was achieved with no degradation in performance or increase in unit cost. Significant cost reductions are possible with the mastic insulation demonstrated.

"This was the first firm fixed price contract awarded under Program 623A for the demonstration test of a large solid propellant motor. Since the contract was tightly negotiated, cost performance became heavily incentivized and firm financial control was exercised by Lockheed management."

The following figure and table summarize LPC's related program experience.



LPC SRM EXPERIENCE

120-INCH	TEST DATE	PROPELLANT WT. (K-LB)	THRUST (M-LB)	BURN TIME (SEC)	PROPELLANT TYPE	TVC TYPE
120" ARM	5-12-62	163	0.3014	122	PBAA (543A)	SITVC (Freon/H_2O_4)
156-INCH						
L-71	5-28-64	423	0.9486	108	PBAA (543B)	JET TAB
L-72	9-30-64	626	1.101	142.8	PBAA (543D)	JET TAB
156-5	12-14-65	687	2.84	55.25	PBAN (580A)	LITVC (N_2O_4)
156-6	1-15-66	278	0.964	65	PBAN (580C)	LITVC (N_2O_4)
HGV	4-7-66	156	0.2718	121.7	PBAA (592A)	SITVC (Hot Gas)

Lockheed Propulsion Company has also achieved significant advancements in state-of-the-art technology through design and development of the propulsion subsystem for the AGM-69A Short Range Attack Missile (SRAM). This high-performance, two-pulse solid propellant rocket motor incorporates design features unique to pulse motor technology, including a thermal-barrier-between-pulse concept, a grain retention system capable of withstanding severe motor pressurization and environmental loads, and insulative materials especially developed by LPC to withstand the heat soak between pulses. SRAM is the first high-burn-rate, end-burning, solid propellant pulse motor ever developed for air-launch missile application.

Substantial technical achievements were also realized in development of SRAM components suitable for high-pressure, long-duration operation. These include the high-performance D6AC steel case, the high pressure and pulse capability of the nozzle, the high-burn-rate propellant with its wide temperature capability, the superior case insulation system, the ignition system with its on-demand pulse capability and high-energy initiators, and the unique pressure equalization and interpulse barrier features of the grain retention system.

Successful completion of qualification testing of the highly sophisticated SRAM propulsion system in March 1971 culminated a $4^1/_2$ year DDT&E program and represents an outstanding achievement in solid rocket motor technology. This achievement was followed immediately by the equally successful introduction of the SRAM motor into full production. Here again, LPC technological capability and management know-how successfully overcame the new challenges encountered during conversion from R&D. The first production contract was awarded to LPC in July 1970 and, despite many problems, the first production unit was delivered ahead of schedule in October 1971. Since then, the build-up to full production rate has remained on or ahead of schedule. The SRAM missile became operational on 1 March 1972.

Lockheed Propulsion Company has served NASA space programs in the development and production of the Mercury Escape Motor and the Apollo Launch Escape and Pitch Control Motors. The Apollo Launch Escape Motor is the largest solid rocket motor ever qualified and used in a man-rated system. Some 70 ground- and flight-firing operations have demonstrated an Escape Motor reliability in excess of the 0.998 specification requirement. Fortunately, the use of this motor in an actual escape maneuver has never been necessary.

Another LPC innovation was the invention of the Lockseal movable nozzle in 1963. This unique, rugged, and reliable flexible joint, consisting of elastomeric pads and structural reinforcements laminated into a composite structure, was first demonstrated in a thrust vector control (TVC) system in 1964. Since then, it has been used in numerous applications both in and outside the aerospace industry. More than 800 Lockseals have been produced to date on 40 separate design, demonstration, special test, and production contracts. Lockseals are used in both the first- and second-stage TVC systems of Poseidon production motors and have performed excellently in this application.

The extensive technological experience gained on these and other LPC rocket motor and related programs was applied to the current SRM study and is reflected in the results described in this report.

Section 9

SUPPORTING DOCUMENTATION

Supporting documentation previously provided to the National Aeronautics and Space Administration is listed below:

Document No.	Title	Description	Date
629-2	Environmental Impact Statement for SRM Booster in Conjunction with the Space Shuttle Program	Environmental Analysis of SRM application	14 Feb 72
629-3	Vol I Program Review	Presentation data	14 Feb 72
629-3	Vol II Program Review Supporting Data	Detailed data supporting the Program Review	14 Feb 72
629-3	Viewgraphs	One set of program viewgraphs	14 Feb 72
629-4	Study of the Reuse of SRM Booster Vehicles	Recovery and reuse presentation data	Not dated (provided on 2-14-72)
OKL 20301	Minutes of Briefing	Minutes of 14 Feb 72 Program Review	18 Feb 72
629-5	Executive Summary	Presentation data	23 Feb 72
629-5	Viewgraphs	One set of viewgraphs of the Executive Program Review	23 Feb 72
OKL 20391	Minutes of Briefing	Minutes of 23 Feb 72 Program Review	8 Mar 72

Appendix A

EXTERNAL LOADS ON VEHICLE
(156-Inch SRM, Parallel-Burn, Heavy Payload)

A.1 LOAD ASSUMPTIONS (Figure A-1)

(1) No significant transportation or assembly loads.

(2) On launch pad prior to launch, 99-percent wind is approximately 50 knots.

(a) Vortex shedding load factor on above wind is equal to or less than 10.

(3) Skirts are clamped to launch stand and orbiter motors ignite and build up thrust quickly.

(a) When orbiter ignition is verified, clamps are released and SRMs ignite simultaneously.

(b) SRM motor growth of 1.05 inch at ignition is too slow to accelerate orbiter significantly.

(c) Orbiter ignition and dynamic overshoot of structure cause skirt loads and connector loads.

(4) Abort may be required if there is excess chamber pressure and nozzle is swivelled to any position.

(a) Excess chamber pressure assumed to be 1.5 times nominal.

(b) Nozzle swivelling is in conjunction with zero and inertia loads, but the correlation is loose and loads used correspond to static nozzle forces only.

(c) Thrust termination is sudden and causes dynamic reversal of above forces existing just before termination.

(5) If vehicle has enough altitude at abort, motors separate by hinge mechanism.

(a) Snubber overshoot factor is approximately 1.2.

(b) One worst-load condition is with full propellant weight and slow speed.

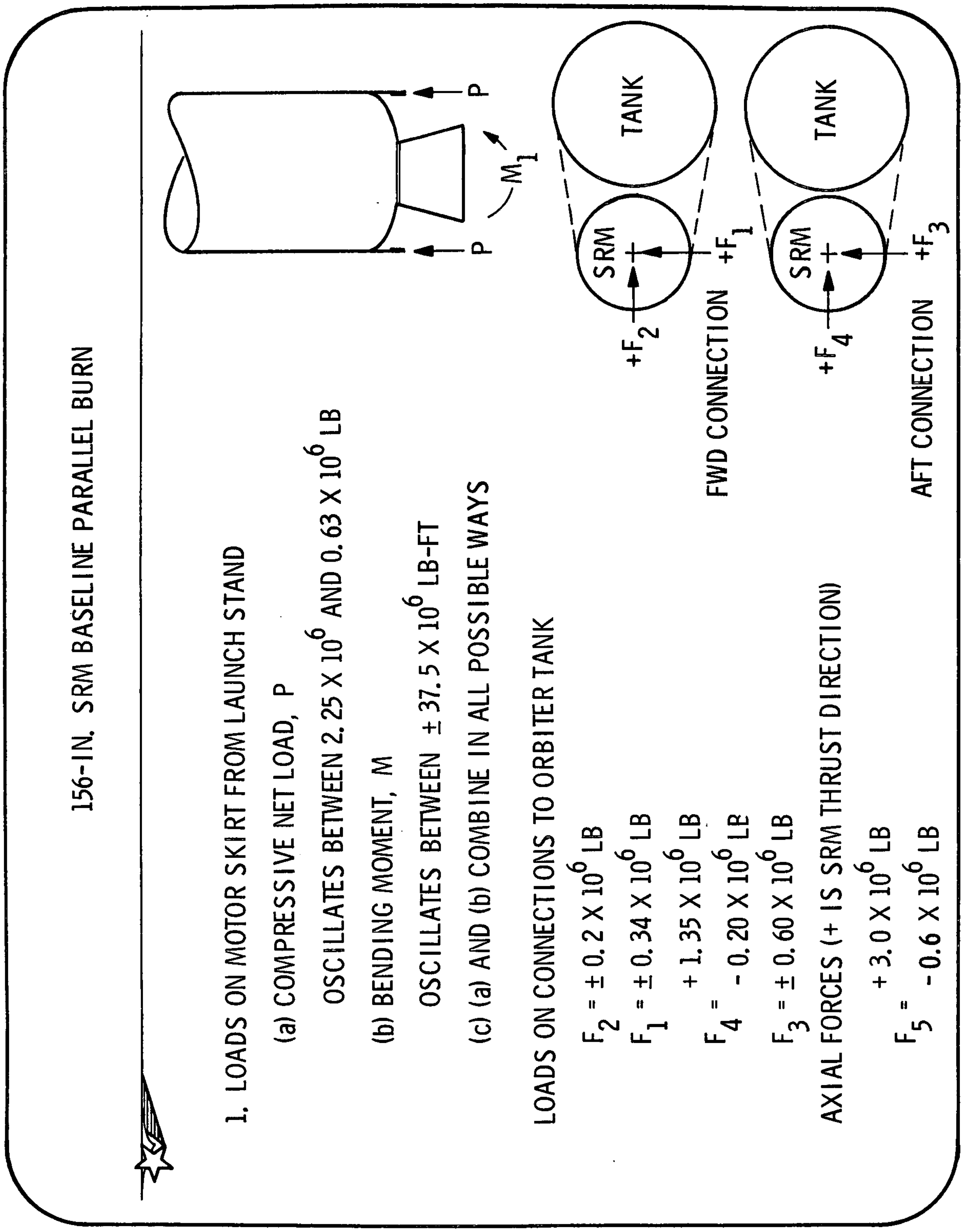


Figure A-1 Load Summary for Baseline 156-Inch Parallel-Burn SRM with Heavy Payload

(c) Another may be around maximum $\bar{q}$.

(6) Loads are limit loads. Ultimate loads are 1.5 times limit.

A.2 WIND LOADS ON VEHICLE ON LAUNCH STAND

Per NASA Technical Memo Report 52872, page 5.228, "Recommended Wind Criteria":

(1) Used wind steady velocity distribution with height above ground and 10-minute gust factor to compute steady force on vehicle.

(2) Used strong wind speed spectra to compute response to randomly varying winds.

(3) Results gave vehicle-to-launch stand limit loads.

A.3 IN-FLIGHT WIND AND GUST LOADS

Per above NASA Memo:

(1) Used wind-versus-altitude profile to determine angle of attack at altitude of maximum aero-pressure.

(2) Calculated resultant aero-load distribution and nozzle angles required to control vehicle.

(3) Calculated distribution of lateral acceleration and inertia loads resulting from (2) above.

(4) Calculated shears and bending moments in interstage due to (1), (2), and (3) above.

(5) Increased loads to account for added gust.

A.4 SRM NOSE FAIRING LOADS

Shear, bending moment, and axial load for the SRM nose fairing are presented in Figure A-2. Two conditions are presented for axial load, namely:

<u>Condition 1</u> - Maximum axial compression when combined with bending moment results in maximum compression line load.

<u>Condition 2</u> - Minimum axial compression when combined with bending moment results in maximum tension line load.

Table A-1 presents the maximum pressure differentials acting on the nose fairing at the time of maximum dynamic pressure.

Table A-1

SRM FAIRING PRESSURE DIFFERENTIALS
AT MAXIMUM DYNAMIC PRESSURE

Hemisphere nose:	maximum crushing pressure	= 10 psid
Forward cone:	maximum crushing pressure	= 6.2 psid
Aft cone:	maximum crushing pressure	= 3.5 psid
	maximum bursting pressure	= 2.0 psid

NOTE: Maximum burst pressure at any time prior to maximum $\bar{q}$ is not expected to exceed 5 psid.

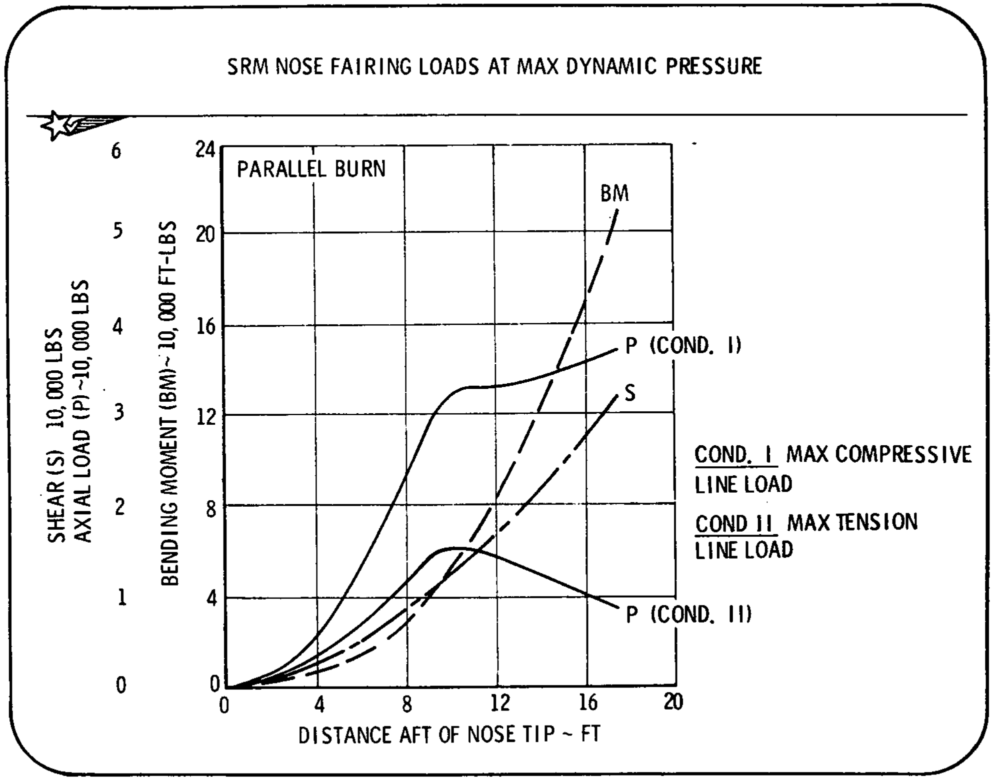


Figure A-2 SRM Nose Fairing Loadset Maximum Pressure, Parallel-burn SRM

Appendix B

ENVIRONMENTAL IMPACT STATEMENT FOR SRM BOOSTERS IN CONJUNCTION WITH THE SPACE SHUTTLE PROGRAM

ENVIRONMENTAL IMPACT STATEMENT FOR SRM BOOSTER IN CONJUNCTION WITH THE SPACE SHUTTLE PROGRAM

SUMMARY

Available solid rocket motor (SRM) data and related environmental analyses were evaluated in conjunction with application of SRM boosters in the NASA Space Shuttle Program. Potential environmental impact of this program is summarized in Table I. No significant overall impact is expected from the development of such boosters or their use in the Space Shuttle Program.

The contribution of the SRM boosters to environmental pollution appears to be many orders of magnitude below those of other sources when assessed in terms of global or national significance. The immediate launch pad and static test areas are subjected to noise levels equivalent to Saturn V launches and to potential momentary effects of aborts. The combination of events leading to accidents or flight failures has yet to be experienced during the Titan vehicle programs which use large SRM boosters. The probability of confining launch products to the controlled test area is believed to be very high.

Launch constraints for the environmental aspects of the Space Shuttle vehicle do not appear to be a requirement; under worst conditions, constraints would not exceed those associated with Saturn V. LPC is not in a position to evaluate other possible constraints that could be more severe than those for Apollo (i.e., crew visibility requirements for an abort and fly-back sequence). The effects of such launch criteria are anticipated to be faborable for assimilation and dispersal of the plume cloud formed at the launch pad during liftoff and the on-trajectory plume without environmental impact.

Previous Space Science and Applications Spacecraft launched by NASA vehicles have made significant contributions to understanding and use of the environment. The Space Shuttle Program, using SRM boosters, is expected to continue this history of benefits; e.g., improvement and protection of the ecosystem which are considerations in harmony with the overall objectives.

In the commitment of resources to this program, raw materials in the propellants and launch vehicle can be considered, in the practical sense, to be irreversible and irretrievable. These materials are easily replaced, and are insignificant, for example, in comparison with resources and energy required to produce the current production rate of 1,000,000 barrels of jet fuel per week. The majority of program costs is wages and salaries representing a small fraction of the National economy. Consequently, commitment of resources to this program is expected to have a small, but positive effect on the National economy.

TABLE I

SUMMARY OF ENVIRONMENTAL IMPACT OF SRM USE IN THE SPACE SHUTTLE PROGRAM

Area of Concern	Propellant Processing	Static Test	Normal Launch	Accident or Abort
Air pollution	No significant effect with development of incineration disposal technique	Local impact possible under combination of adverse meterological conditions and HCl concentration in plume	Local impact possible under combined adverse meterological conditions and plume HCl concentration, SATURN V launch criteria mitigates this potential	Limited area possibly subjected to HCl concentrations above exposure criteria in event of an on-pad fire or low-level abort
Water pollution	No significant effect with primary on-site treatment	No significant effect	No significant effect	No significant effect
Noise	No significant effect	Test area impact	Test area impact. SATURN V equivalent	Test area impact. SATURN V equivalent
Re-entry debris	N/A	N/A	No significant effect	No significant effect
Environmental enhancement	No significant effect	Positive contribution in context of total program	Space Science and Applications Spacecraft make positive contributions	No significant effect
Commitment of resources	No significant commitment of limited resources	No significant commitment of limited resources	No significant commitment of limited resources	No significant commitment of limited resources

NASA SPACE SHUTTLE PROGRAM DESCRIPTION

A space shuttle system, utilizing an expendable booster and reusable orbiter that can transport persons and cargo to low earth orbit and return the crew/passengers and cargo safely to earth, is now in the definition stage of development (Phase B). A secondary effort will be applied to the definition of recovery and reuse of solid rocket motor parts for the purpose of cost comparison with other reusable systems. The objective of this effort is to establish a specific design, a development program, a production program, launch operations, vehicle support, and definitions for SRM-propelled Space Shuttle.

The initial SRM development, processing, and static test ballistic evaluations will be performed at Lockheed Propulsion Company's Potrero facility located near Beaumont, California. Launch-phase SRM processing will be conducted at, or in the vicinity of Kennedy Space Center launch site, in Florida. The general program plan for SRM development and production to support the launch rate is shown in Figure 1.

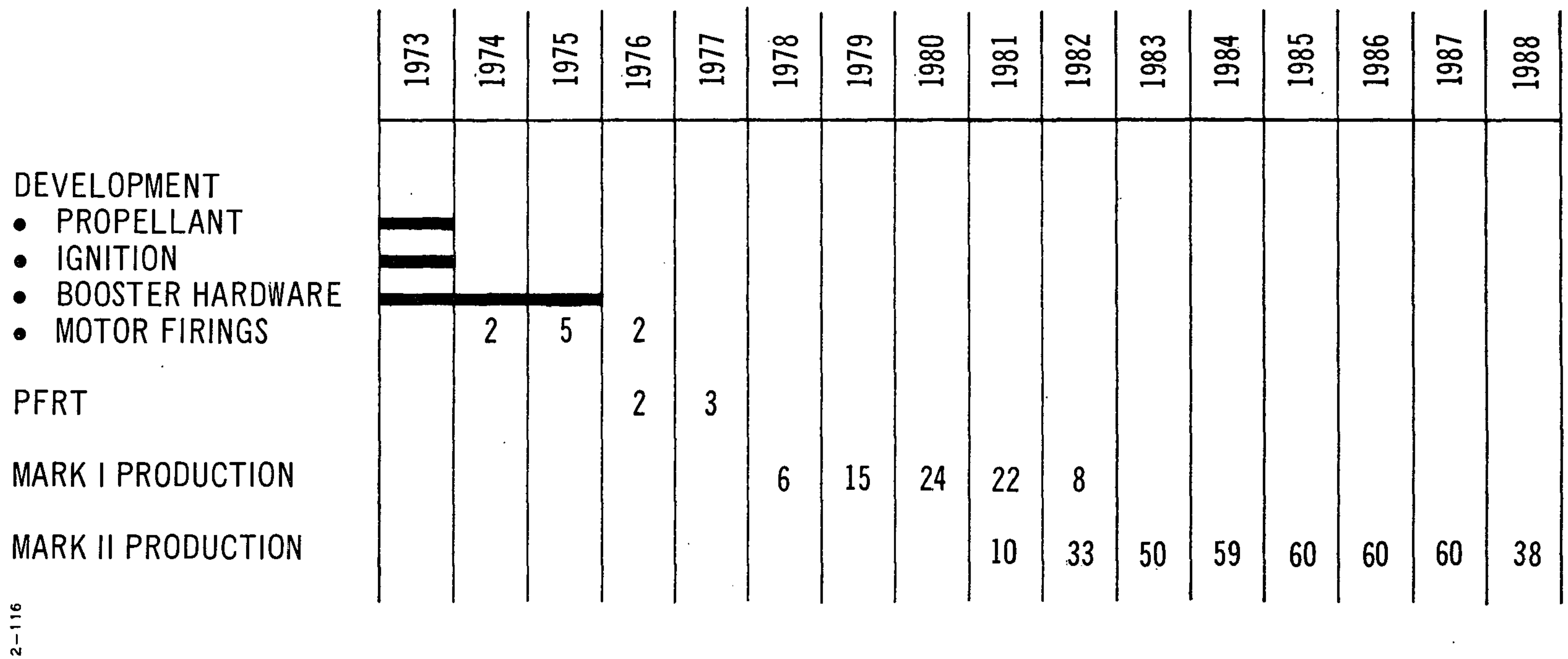


Figure 1. - Approximate Program Plan, Solid Rocket Motor for Space Shuttle Booster

ENVIRONMENTAL IMPACT OF PROGRAM ACTIONS

The Space Shuttle Program using an SRM booster has been investigated for possible environmental impacts in the following areas:

- Population shifts (due to manpower needs)
- Air pollution
- Water pollution
- Reentry of launch vehicle debris
- Noise

The major activities will be concentrated in, but not restricted to, Southern California and Florida. No significant population shifts are expected during the performance of the program.

SRM advanced studies, most research and development activities, manufacturing, and most testing are relatively clean and quiet operations, and will not directly produce significant environmental effects. However, such activities will consume power, metals, paper, and other materials that may induce secondary impact on the environment. This secondary impact is difficult to quantify, but is not likely to vary from the consumption levels being experienced in support of continuing aerospace activity. Consequently, it is dismissed from further consideration.

Research, development, and testing of rocket propulsion systems result in the handling and consumption of relatively small amounts of propellants. These programs may contribute to air and water pollution and noise generation. Currently, acceptance testing of production liquid propellant rocket engines is the major consumer of propellants in research and development. This ratio between liquid propellant and solid propellant testing will obviously change as a result of the selection of an SRM booster for the Space Shuttle. The impact of these activities is presented in the subsequent sections.

The actual launch and flight of Space Shuttle vehicles is the major activity affecting the environment. In addition to normal vehicle flight, the impact of possible abnormal flight conditions will be considered in the following paragraphs. It should be noted that the preparations for all launches include an extensive safety analysis for both normal and possible abnormal events. The vehicle trajectory, flight sequence, launch date and time, and other parameters are adjusted as necessary to meet safety requirements.

Air Pollution Assessment

SRM boosters are chemical rockets that produce thrust by the combustion of a fuel and self-contained oxidizer. The products of combustion exhausted from the rocket nozzle may include compounds and molecular fragments which are not stable at ambient conditions, or which may react with the ambient atmosphere. Knowledge of the detailed composition of rocket exhaust gases is largely based on thermochemical calculations, which assume that the propellants are completely mixed in the combustion chamber. If available, actual measurements of the gas composition, especially after discharge from the nozzle and mixing with the ambient atmosphere, would be a preferred source of information on the composition. In one of the two primary Space Shuttle vehicle configurations, the operation of the SRM boosters will be augmented by Shuttle Orbiter rocket operation. The concentration of combustion species in the plume will be different for the two configurations, but there will be no change in the basic list of species encountered.

Identification of substances emitted by rocket engines may be derived from the nominal propellant, from additives in the propellant, from impurities in the propellant, or from the insulation or other components that ablate in the hot gas flow. Major chemical species emitted by the SRM and Space Orbiter are listed below:

Water	Nitrogen
Carbon dioxide	Hydrogen
Carbon monoxide	Aluminum oxide
Hydrogen chloride	

Minor constituents (in terms of concentration) may include:

Carbonaceous particles (smoke)	
Aluminum chloride	Silica
Nitric oxide	Iron oxide
Sulfur dioxide	Normal hydrocarbon fragments

Of the major constituents, carbon monoxide (CO), and hydrogen chloride (HCl), are generally recognized as air pollutants presenting a toxicity hazard. Aluminum oxide (Al_2O_3), would be classified as an "Inert or Nuisance Particulate" by the American Conference of Governmental Industrial Hygienists. In the upper atmosphere, water (H_2O), and carbon dioxide (CO_2), may be considered as potential pollutants due to their low natural concentration. The following discussions contain more detailed investigations of air pollution impact from exhaust species.

Basic exhaust specie distribution. - The exhaust specie concentrations for the series Space Shuttle Vehicle configuration is that of the SRM boosters operating alone. In the parallel configuration the Orbiter hydrogen/oxygen (HO) engine will be operating simultaneously with two SRM booster units. The nozzle exit specie concentrations for the SRM booster are listed in Table II (values in the table are weight percent). The Shuttle Orbiter exhaust is virtually 100 percent H_2O including dissociated fractions.

TABLE II

SPECIE CONCENTRATION

Specie (weight-percent)	SRM Exit Plane	SRM Plume	SRM and Orbiter Plume
CO	20.8		
CO_2	3.3	24	16
Cl	0.3		
HCl	21.2	21	13
H	0.03		
OH	0.03		
H_2	1.9		
H_2O	9.9	12	41
N_2	8.4	9	6
$AlCl_2$	0.04		
Al_2O_3 (solid)	34.1	34 + air	24 + air

Table II also contains estimates of the plume specie concentration after atmospheric processes are completed, such as afterburning of fuel species with ambient oxygen. The plume specie entries indicate the probable conversion of CO to CO_2 by oxidation in high temperature afterburning, and conver- of H_2 into H_2O. Numerous analytical and experimental rocket plume investigations support this contention. As a result, CO may be expected to be a very short lived pollutant of minor significance. The SRM and Orbiter plume data includes the combination of two SRM plumes and the Orbiter plume in direct proportion to the mass flow rates from each rocket.

Inventory of related emissions from other sources. - A general inventory of emissions related to the exhaust species of the SRM booster propellants as derived from various other sources is listed in Table III. NASA launch vehicles in use during 1969 to 1971 are included for reference and as a basis for comparison to SRM booster data. While NOx is a minor specie from chamber combustion processes (0.0015 weight-percent) it is a potential product of propellant burning in atmospheric conditions with excesses of air present. Such might be the case in an SRM booster accident in the launch pad area. The CO value for NASA launches is believed based on chamber values, and not plume values following afterburning.

Exposure criteria. - A summary of exposure criteria for rocket combustion products is shown in Table IV. A variety of conditions associated with the exposure to HCl and CO are listed. A distinction is made between uncontrolled and controlled populations. Personnel safety considerations surrounding static tests and launches always yield carefully controlled populations.

Normal launches. - The first few seconds of a normal sequence are consumed in systems checkout with the vehicle restrained, liftoff, and low velocity vehicle rise. Exhaust products are momentarily diverted into a flame bucket

TABLE III

INVENTORY OF EMISSIONS INTO THE LOWER ATMOSPHERE

Source	Emission, 10^6 lb/year					
	CO[f]	NO_x	HCl	SO_2	Ash	Urban Particulate
NASA Office of Space Science and Applications Launch Vehicles, 1969-1971 average[a]	0.912	0.0003	0.133	---	0.239[d]	
Automobiles[b]	124,000	12,600	9[c]	---	---	30 to 90[e] tons/sq mile/mo
Power plants[b]	200	7,000	1340[c]	29,600	9,800	
Trash incineration[b]	15,200	1,000	400[c]	---	---	
Jet aircraft[b]	600	200	---	---	---	
SRM static tests (average yearly rate)	0.8	0.006	1.02	---	1.52[d]	
Launches, 6/yr	0.9	0.007	1.15	---	1.71	
Launches, 60/yr	9.0	0.07	11.5	---	17.1	

(a) Based on first stage propellants

(b) For 1966. Sources: APCO/DAQED, DPCE and APCO(NAPCA) reports

(c) Estimates from Gerstle and Devitt, "Chlorine and Hydrogen Chloride Emissions and Their Control," Paper No. 71-25, Air Pollution Control Association, 1971

(d) Al_2O_3 from solid propellants

(e) All sources, "Solid Rocket Motor Interim Booster Concepts" McDonnell-Douglas Astronautics Company, Design Note B-W-TI-10, October 1971

(f) In afterburning processes of the rocket plumes with ambient air, conversion to CO_2 is virtually assured.

TABLE IV

EXPOSURE CRITERIA FOR COMBUSTION PRODUCTS

Substance	TLV[a]	Maximum Concentration Short-Term Industrial Exposure[a]	Suggested Short-Term Emergency Limits[b]	Uncontrolled Population
HCl	5 ppm	5 ppm	20 ppm, 30 min; 10 ppm, 1 hr	3 ppm, 10 min; 1 ppm, 30 min[d]
CO	50 ppm	75 ppm	200 ppm, 1 hr	30 ppm, 1 hr[e]
Al_2O_3	15 mg/M^3[f]	-----	-----	-----
$AlCl_3$	10 mg/m^3[c]	10 mg/m^3[c]	-----	-----

(a) From "Threshold Limit Values of Airborne Contaminants", American Conference of Governmental Industrial Hygienists, 1970; TLV's are based on a time-weighted exposure for 40-hr work weeks.

(b) From "Compendium of Human Responses to the Aerospace Environment", Vol. III, NASA CR-1205(111), November, 1958

(c) Based on hydrolysis to HCl. In subsequent discussion, $AlCl_3$ is considered only in terms of its contribution to overall HCl levels.

(d) From Cramer, H. E., et al., "Titan-IIID Toxicity Study", GCA Report No. TR-70-3-A, June, 1970

(e) Based on 1.5 percent Carboxyhemoglobin in 1-hr exposure. See "Air Quality Criteria for Carbon Monoxide", U.S. Department of Health, Education and Welfare, March 1970, Publication AP-62.

(f) Basically a particulate standard

and through fluid mechanics effects, form an approximately symmetrical plume cloud about the base of the vehicle. In approximately 10 seconds of operation, the vehicle achieves sufficient altitude that the mass flow from rocket exhausts no longer enters the plume cloud. A recent paper by Hart (ref. 1) describes the formation, rise, and growth of plume clouds for SRM booster Titan vehicles.

The remainder of the exhaust products are distributed along the vehicle trajectory. Due to the acceleration of the vehicle and the staging process, the quantities per unit length of trajectory are greatest at ground level, and decrease continuously. The environmental impact of exhaust products distributed along the trajectory will be discussed separately as lower and upper atmosphere effects. The launch plume cloud occurs at the lowest atmospheric level and will be discussed first.

The plume cloud rise and growth estimated by scaling Titan vehicle data are shown in Figure 2. The initial plume radius is 590 feet and is presumed fully formed at t = 35 seconds, at which time the entire plume cloud rises from the launch pad. The simultaneous growth in radius and gain in altitude can be seen in Figure 2. At the time the plume cloud rises, an estimated bulk temperature of 580^0R exists, and the cloud has entrained approximately 250 times the weight of propellant exhaust in air along with atmospheric moisture. After $2\frac{1}{2}$ minutes, the plume cloud has attained a radius of approximately 935 feet due to expansion and, depending upon meterological conditions, an altitude (center of plume) of nearly 2500 feet.

Plume cloud shape is preserved by thermodynamic and pressure gradient conditions on the interface. Very small quantities of gaseous exhaust products may escape due to diffusional processes as may be judged from the HCl experimental data summary in Figure 3. Figure 3 also shows the HCl input to various layers of the atmosphere as the entire vehicle rises. In estimating the local concentrations of potential pollutants in the plume cloud, it is assumed that the plume cloud boundary contains all exhaust products and that the majority of CO and H_2 are oxidized in the afterburning process. Therefore HCl is the primary concern for local impact. The plume cloud will contain most HCl when the series configuration is launched. Under this condition, the initial concentration of HCl gas is approximately 810 ppm, including all exhaust gases and the entrained air in the concentration calculation.

As the plume rises and grows, the entrained air quantity increases so that HCl concentration is diluted. This process is illustrated in the left panel of Figure 4. There is, however, water vapor in the plume cloud due to rocket exhaust products, and hydration of the HCl is a potentially fast process (refs. 2 and 3). Estimates of the reduction of HCl gas and hydration to an aqueous aerosol were made using data contained in reference (2). The concentration of HCl with only rocket exhaust water in the plume with time (series configuration) is shown in the left panel of Figure 4. The atmospheric water present in the plume is one order of magnitude greater than the rocket exhaust water, even when the HO Orbiter engine is operating. Consequently, even at low relative humidity, the depletion of HCl as gas is virtually assured.

The analysis shown in the left panel of Figure 4 indicates depletion of HCl gas. Conversion to an aqueous aerosol is investigated in the right panel of the

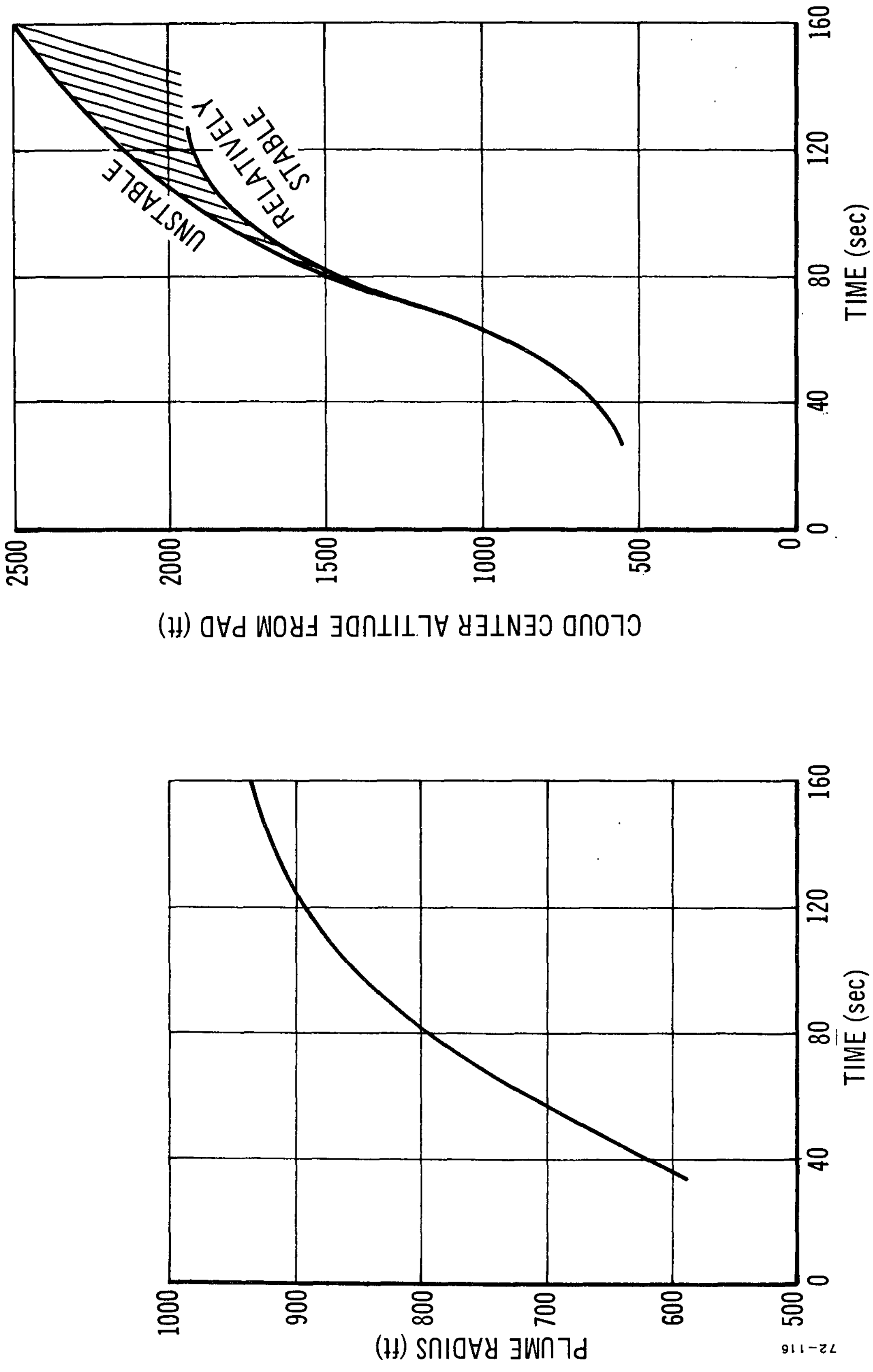
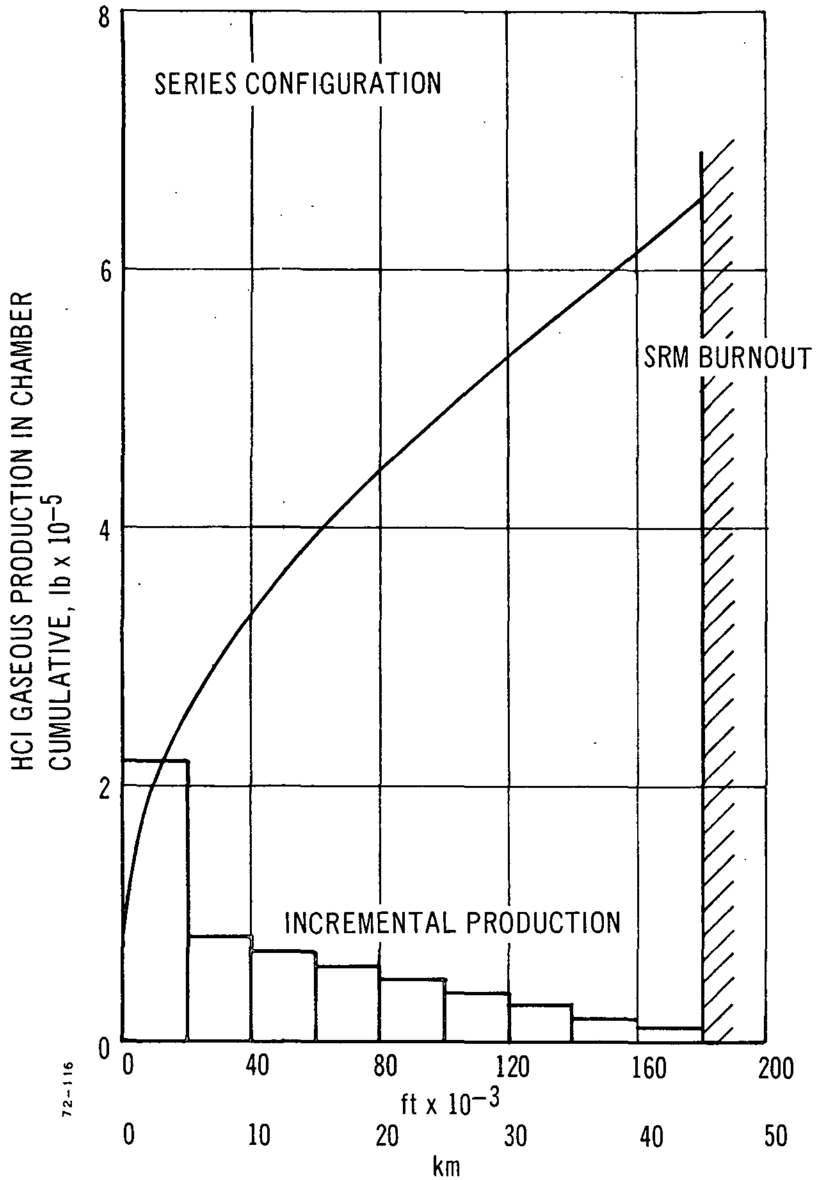


Figure 2. - Environmental Impact Considerations, Plume Cloud



CONCLUSION: NO IONIZATION IMPACT ON EITHER 'D' OR 'E' LAYER OF IONOSPHERE (> 67 km)

SUMMARY, LAUNCH AREA DATA

- FLY THROUGHS, 120 AND 260 INCH STATIC TEST CLOUDS
  0 - 0.8 ppm
- TITAN IIIC, LAUNCH PAD
  r = 600 ft, 0 ppm
  r = 50 ft<3 ppm, t > 45 sec
- MINUTEMAN LAUNCH PAD
  <1 ppm

Figure 3. - Environmental Impact Considerations, HCl Gaseous Production Rate

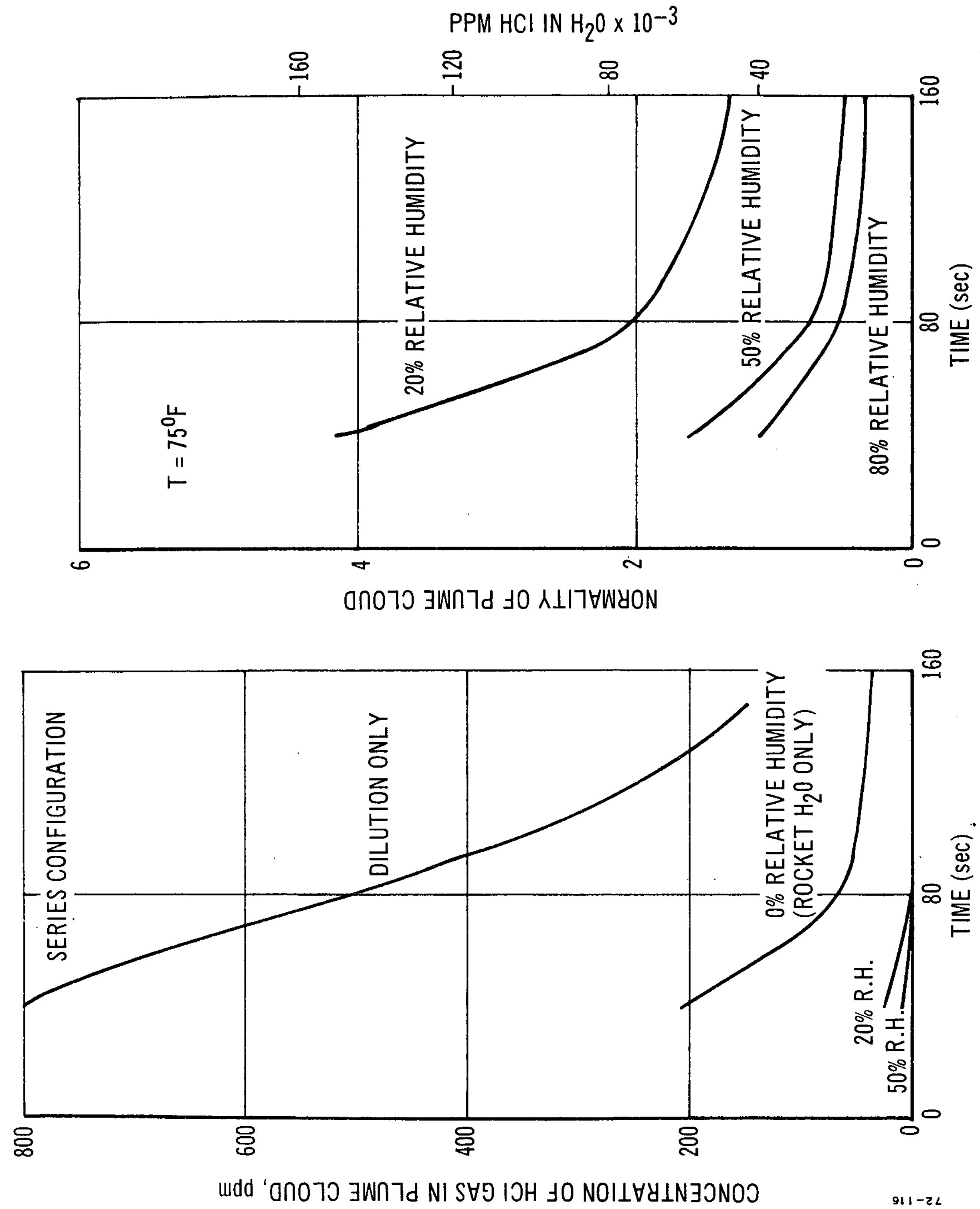


Figure 4. - Environmental Impact Considerations, HCl Dilution and Hydration

figure. Again for the series configuration, the normality of the plume cloud and the concentration, expressed as ppm HCl per part of H_2O in the plume cloud, is plotted as functions of time and relative humidity. The decrease is rapid with time and at high values of relative humidity (common at the ETR) the plume cloud is approximately 0.3 Normal in 2½ minutes. The parallel configuration will yield about ⅔ of the concentrations shown in Figure 4 because only two SRM boosters are used instead of three. The Orbiter HO engine water contribution is small compared to atmospheric water at high relative humidity.

The analysis of potential environmental impact at low altitude during launches can be summarized in the following way:

- The possibility of personnel exposure to HCl is small since HCl will be confined to the plume cloud, which can be tracked visually. Further, depletion of gaseous HCl is likely to be rapid.
- Rapid hydration of HCl to an aqueous aerosol is probable and the concentration may approach a level of 0.3 Normal.
- This degree of concentration will be gradually dispersed by atmospheric turbulence and mixing, provided that local meterological conditions are not conducive to precipitation, i. e. , virtually 100 percent relative humidity with cloud cover and the potential of cooling.
- Under launch constraints currently in effect for Saturn V, which may be tightened by considerations of orbiter crew visibility requirements, the precise conditions for launch area precipitation are not anticipated to occur.
- Prevailing wind conditions at ETR are seaward where an incident of precipitation will be insignificant.
- A precipitation incident over land should be evaluated for potential impact on vegetation as a function of time after launch, degree of plume cloud dispersal, atmospheric mixing potential, and related factors.

Environmental impact of a very selected nature has been identified, however, the occurrence of specific conditions is rare and may be further reduced by crew and system launch constraints unrelated to propulsion.

Environmental impact of CO_2 generated in the lower atmosphere through afterburning is shown in Figure 5. A representative worst case selection was made to illustrate CO_2 impact. A static test firing, which releases virtually all the exhaust products in lower atmosphere levels, provides 2.65×10^5 lb of CO_2, presuming all CO is oxidized. Figure 5 indicates CO_2 generation at an hourly rate for fossil fueled power plants of various capacity. A 300-megawatt power plant operating for 1 hour at full capacity generates an equivalent amount of CO_2 as a full static test of an SRM booster.

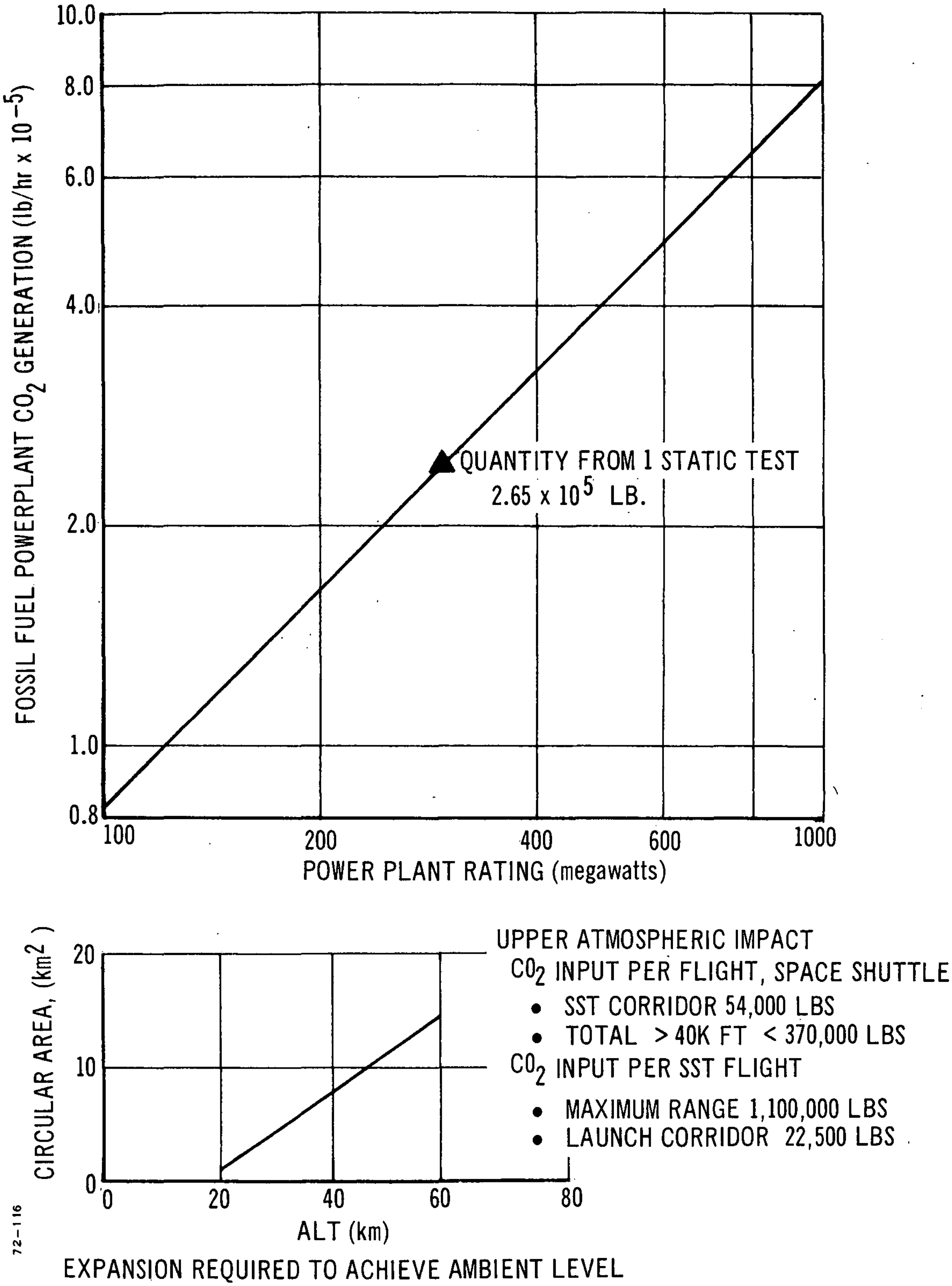


Figure 5. - Environmental Impact Considerations, CO_2 Generated in the Lower Atmosphere

The final lower atmospheric effect is particulate deposition of Al_2O_3. While a large fraction of the Al_2O_3 is generated at high altitudes, eventual settling of all the Al_2O_3 to the ground was presumed. The average value over the launch corridor of 2 by 70 miles as a function of launch rate is shown in Figure 6. Even at high launch rates of 40 per year, the deposition of Al_2O_3 does not approach typical urban particulate fallout rates.

Investigation of the upper atmospheric effects shows the parallel configuration vehicle, with the HO Orbiter engine operating, emits the largest amount of water in the stratospheric layer. An estimate was made of the exhaust cloud spread that would be required before the H_2O concentration fell to the ambient value as given in the U. S. Standard Atmosphere. At a 25 km altitude, the effects of the cloud would blend into the ambient background by the time it expands to 6.5 square kilometers. (Series configuration requires only 2.3 square kilometers.) At 45 km altitude or approximately at the point of SRM booster burnout, the cloud would have to expand to 2970 square kilometers to reach equilibrium with ambient H_2O concentrations. (Series configuration requires 1080 square km.)

In the layer above 105 km, Kellogg (ref. 4) has calculated the required number of SRM firings of "nominal super rockets" of the Saturn category which would be required to double the concentration of H_2O, CO_2, and NO in the atmosphere above 105 km. A nominal super rocket is defined as one that ejects 200,000 lb of exhaust into the heterosphere above 105 km. His results indicate that the following number of rocket firings per year would double the concentrations in the heterosphere:

Constituent	Number of Rocket Flights Required to Double the Natural Concentration above 105 km
H_2O	6.7×10^3
CO_2	1.4×10^5
NO	6.5×10^5

Liquid-propellant systems add about three times as much water vapor per pound of exhaust as do the solid propellant systems. Kellogg commented on this additional contribution by stating that the water would be dissociated and the excess hydrogen would escape from the top of the atmosphere after a residence time of only a few days.

The effect of water vapor from a launch vehicle upon the ozone concentration can be considered as negligible from the small area covered by the exhaust cloud. The rocket can create a small hole in the ozone layer but the photochemical processes taking place in the atmosphere will quickly fill up any void of ozone.

Estimates of the area in the stratosphere into which the plume would have to expand before the carbon dioxide density would reach that of the ambient air were made as in the case of water vapor. For CO_2 at 25 km altitude the cloud must expand to less than 0.23 square kilometers before the CO_2 would reach

Al_2O_3 PARTICULATE DEPOSITION

EXHAUST CONTENT OF AL_2O_3 - 34.1%

STATIC TEST
260 INCH DATA:
0.3 - 0.6 TONS/SQUARE MILE
5 MILES DOWNSTREAM
156-INCH SRM ESTIMATE:
0.2 - 0.4 TONS/SQUARE MILE
5 MILES DOWNSTREAM
URBAN PARTICULATE FALLOUT:
30-90 TONS/SQUARE MILE/MONTH

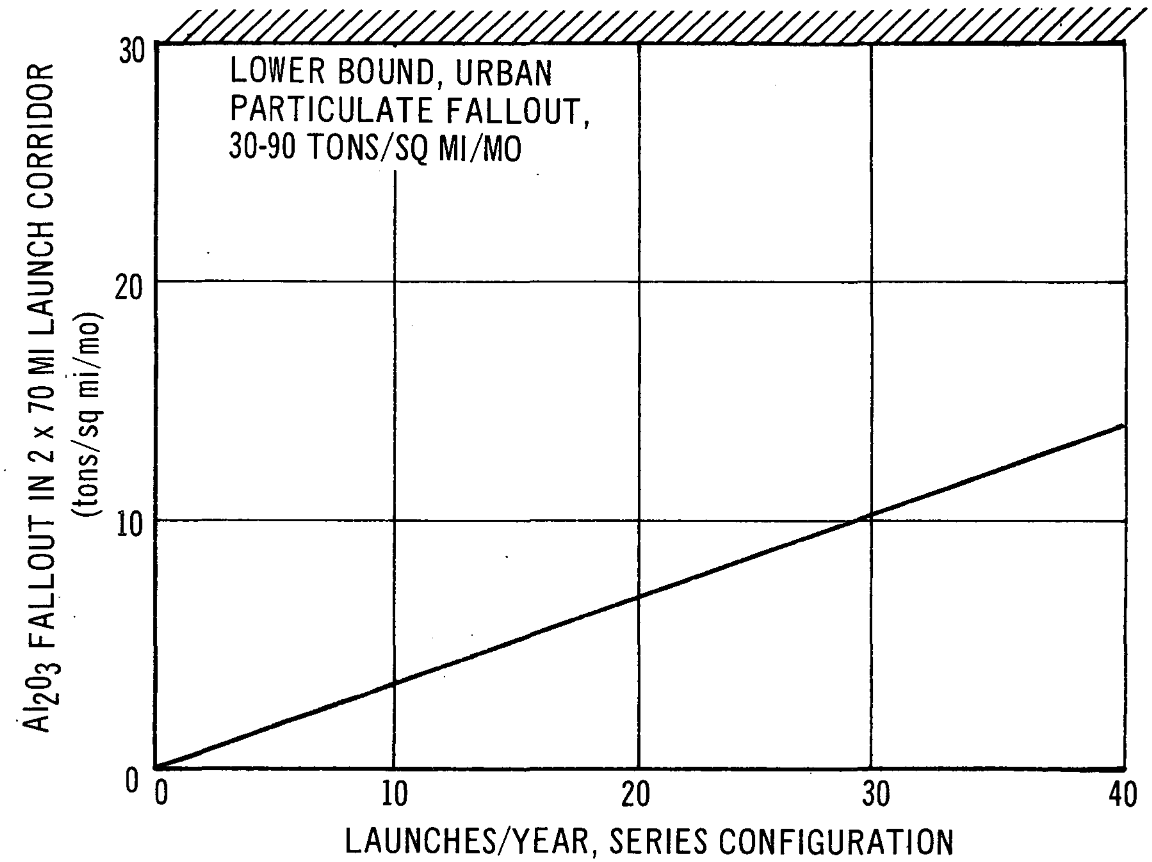


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Figure 6. - Environmental Impact Considerations, Al_2O_3 Particulate Deposition

ambient levels. At 45 km the cloud would drop below ambient levels of CO_2 concentration after it expanded to an area of 5 square kilometers.

The principal concern regarding large increases of CO_2 and H_2O in the upper atmosphere, and above it, are the effects these constituents might have on the global radiation balance, through absorption or scattering of incoming or outgoing radiation. The above estimates of the area required for diffusion of H_2O and CO_2 to background levels indicate the generation of these compounds will have negligible effects.

Calculations of natural NO levels in the layers above 60 km have been made by Barth (ref. 5) in which he predicted concentrations as high as 10^{-2} ppm. The NO emitted from the exhaust of the series configuration SRM boosters (worst case) dissipates below the 10^{-2} ppm concentration when the exhaust cloud expands beyond 10.2 km^2 at 25 km and beyond 770 km^2 at 45 km. It is reasonable to suppose that NO levels above the natural equilibrium level will be reduced through dissociation by solar ultraviolet radiation until the natural equilibrium is again restored.

Hydrogen chloride emissions could have an effect on the ionization level in the upper atmosphere. If this change in ionization level is to have an effect on radio wave transmission (the only effect known to be of importance), the emission of HCl in layers above approximately 90 km (the nominal base of the E layer of the ionosphere) would have to be significant. The SRM burnout altitude is far below this station and could not affect the E layer or the D layer below it.

In summary, there is no significant effect of Space Shuttle SRM boosters on the upper atmosphere. Current activities appear to be many orders of magnitude below those which would be expected to produce detectable changes in the upper atmosphere.

<u>Static tests</u>. - Static SRM tests differ from launches in that all of the propellant used is consumed at ground level. However, the high temperature of the exhaust gases causes them to rise in a buoyant plume. The downwind concentrations of the exhaust gages are critically dependent on the height of this buoyant rise, and any elevation contributed by the persistance of the exhaust jet.

The method suggested by Hage (ref. 6) indicates a buoyant rise of ~500 meters is representative. Using this as a source height, peak downwind concentrations can be estimated by the methods outlined by Turner (ref. 7). The maximum downwind concentration of predicted appears well within suggested exposure limits.

Particulate deposition data are available from previous 250-inch diameter static SRM tests which may be scaled to 156-inch diameter SRM values. The 260 inch static test yielded a particulate deposition of 0.3 to 0.6 tons per square mile at a location of 5 miles downstream of the test. Consequently, the estimate for a 156 inch SRM static test is 0.2 to 0.4 tons per square mile at 5 miles downstream. These values compare favorably with typical urban area particulate fallout values of 30 to 90 tons per square mile per month.

Abnormal launches and accidents. - On-pad accidents involving SRM boosters will produce a situation analogous to static tests with the exception that plume rise will not be aided by jet persistence (exit plane velocity). No occurrence of unintentional on-pad fires involving large SRM boosters are known, therefore making probability estimates of such an event difficult.

Obviously, this lack of samples is encouraging. There are, however, documented intentional large 120 inch SRM destructs, and 156-inch SRM booster static tests from which to judge potential impacts. Analysis of on-pad aborts for the Titan-IIIC vehicle was performed by Barker (ref. 8). Three SRM tests are discussed in this investigation: a 5-segment 120-inch SRM horizontal thrust termination test and a 2-segment destruct test at the United Technology Center, and a 156-inch SRM horizontal static test at Thiokol. In all cases, and although prevailing winds approaching 15 mph existed, the exhaust products rose abruptly. Straight diffusion predictions of up to 5000 feet downwind travel were not observed and the plume cloud shape was fairly well preserved. About 1.4 million pounds of solid propellant was involved in the Thiokol 156-inch diameter test. Barker concluded from experimental evidence and analysis that combustion products will be inhibited from traveling downwind by a stack effect from the heat release.

In the event of an SRM booster failure in flight, crew safety considerations will dictate the responsive action. A worst case possibility might be scattering of burning propellant. The controlled launch pad area appears to be sufficiently large to alleviate momentary effects of this nature.

Based on observation of SRM static tests, therefore, and realistic assessment of heat and atmospheric effects on the Titan IIIC solid motor exhaust products, no significant toxic hazard will result from an on-pad incident.

Waste disposal from processing operations. - SRM booster segment processing will result in solid waste materials and contaminated solvents that must be handled without introducing pollution problems. Figure 7 is a general process flow chart indicating the type and form of waste materials expected. Raw ingredients include oxidizer, polymeric binder material, and aluminum powder. Oxidizer is finely ground to specification for burn rate and solids loading control. The propellant selected for Space Shuttle SRM booster application uses a particularly favorable oxidizer grind that is expected to produce negligible scrap. The weigh-out operations yield contaminated containers and rags.

Uncured bulk propellant is obtained as scrap from propellant mixer residual and many sample containers of propellant are extracted for QA analysis. Tooling cleanup produces contaminated rags, solvent and clean-up aids. Casting operations yield uncured propellant in tooling hang-up and clean-up wastes as with the mixer. The result of QA analysis will be to require repair of a certain number of segments and perhaps even a segment washout. The material from this action may either be cured propellant or spongy residual propellant affected by the washout water jet. In addition, completed aging specifications and milled particles (fines) from sample preparation will be encountered in QA.

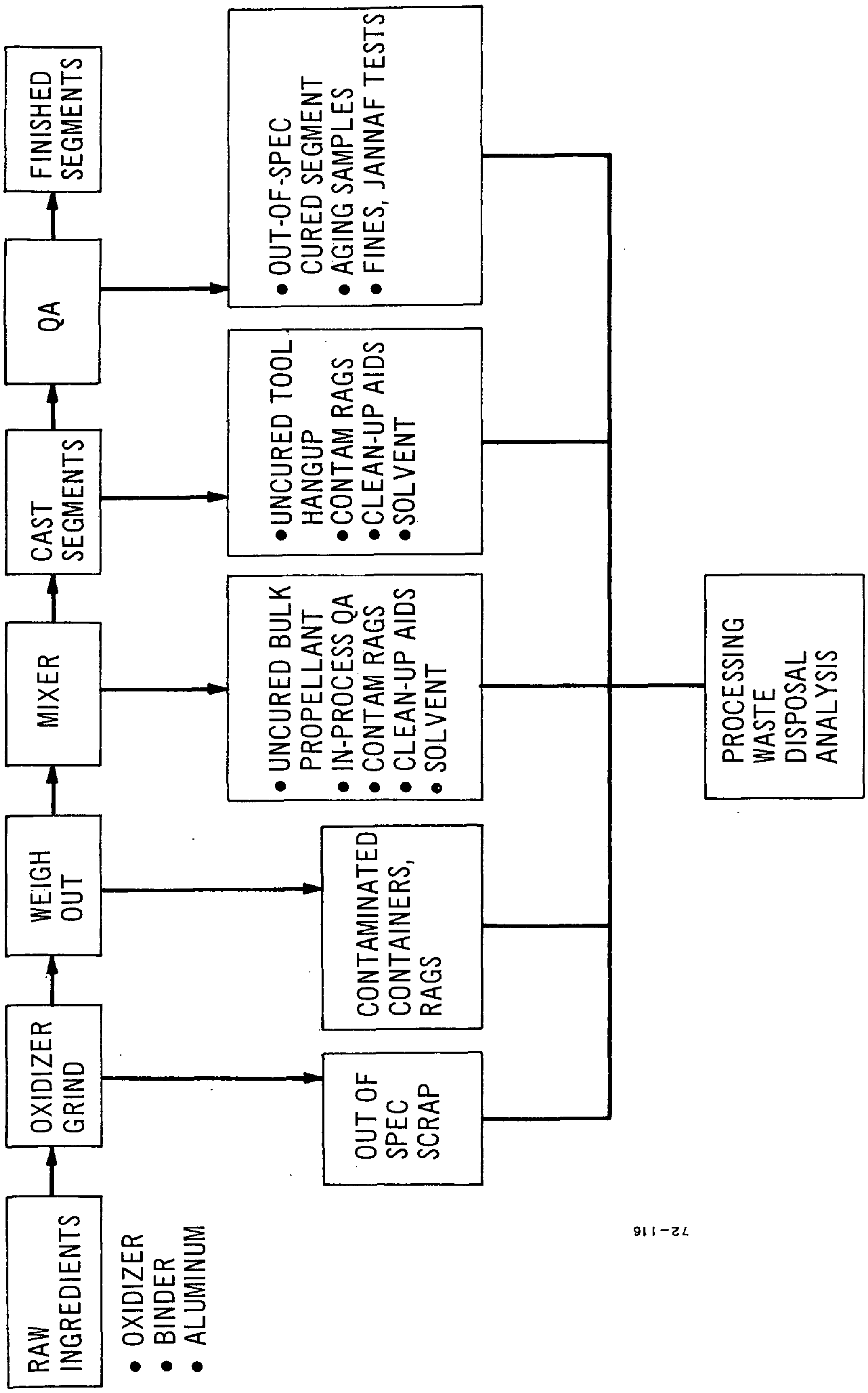

Figure 7. - Process Material Waste

The common industry method for disposing of these hazardous materials in the interest of personnel and public safety has been open burning. However, air pollution abatement efforts in the industry have been initiated in conscientious programs seeking compliance with Federal and local standards. In a recent investigation by Fuller and Taylor (ref. 9), the engineering status of potential air pollution abatement concepts was described along with an industry survey and projections for future implementation. Table V presents a processing waste disposal analysis based on that investigation.

Disposal approaches either currently under investigation or having progressed to pilot plant experiments include transportation to Federal disposal sites, reclamation, on-site incineration, controlled open burning, unique propellant destruction approaches, and ocean dump. Qualitative and quantitative (highes score = 10) environmental ratings led to the selections shown in the right-hand column of Table V.

Reclamation was selected for solvent and oxidizer recovery (if needed). A saturated solvent, 1, 1, 1-trichlorethane, was selected as an accpetable non-toxic, non-participant in photochemical kinetic reaction schemes. Potential relamation methods of ammonium perchlorate (AP) oxidizer is underway. Solvent use, for example, amounts to approximately 1 gallon per 50 pounds of propellant. For disposal of contaminated processing aids, QA fines, uncured bulk, and cured bulk propellant samples, the choice was on-site incineration. This technique is basically sound in providing compliant emission levels. Engineering design problems must be solved for application, and the residual of the process, Al_2O_3, is to be disposed of as solid waste. The most suitable alternate considered was transportation to a centrally located Federal disposal site. An identified development area is public safety in transport, where mixtures of hazardous contaminants are involved. Approximately 0.5 million pounds of waste must be handled initially.

In the case of rejection of a cured SRM segment that cannot be repaired, the method for disposal is controlled open burning due to safety considerations. The potential combustion products for either on-site incineration or controlled open burning of propellant scrap are shown in Table VI. The mixture ratio values across the top of Table VI are ratios of air used in the calculation to the quantity of air required for stoichiometry with 1 pound of propellant. A value of 1.0 for the mixture ratio as defined above is the stoichiometric point for air and the propellant used. Large values of mixture ratio correspond to large excesses of air in the incineration process.

The concentration of CO decreases rapidly as excess air is introduced. This trend is attributed to oxidation of CO to CO_2. The quantity of CO_2 first increases as more air is added, and then decreases as large excesses of air are encountered. Oxidation of CO to CO_2 accounts for the increase and dilution by excess air for the decrease.

TABLE V

PROCESS WASTE MATERIAL DISPOSAL ANALYSIS

Waste Description	Disposal Approaches	Status of Approach	Environmental Rating		Selection
A. Contaminated processing aids	I. Transportation, Federal disposal site	Available 1975	Public safety in transport	6	B, C → II
					A, D, E, F → III
B. Solvents C. Oxidizer	II. Reclamation	1972 - 1975	EPA acceptable solvent, AP potential under study	10	Alternate I
					G → IV
D. QA fines E. Uncured bulk F. Cured bulk samples	III. On-site incineration	Available 1974	• Compliance with air standards • Conversion to solid • Engineering solution req'd • Basically sound	8	
G. Cured out-of-spec segments	IV. Controlled open burning	Current	• Public and contractor safety advantages • Particulate non-compliance	5	
	V. Unique propellant destruction approaches	Available 1974	• Environmentally effective • Hazards and scale • Questionable cost/effective	4	
	VI. Ocean dump	Current	• Unknown marine effects • Containerization • Public safety	3	

TABLE VI

SUMMARY OF INCINERATED PROPELLANT COMBUSTION PRODUCTS

Weight-Percent of Major Products	Mixture Ratio, Air/Stoichiometric Air for 1-Pound of Propellant					
	0.81	1.0*	1.21	4.56	8.11	71.7
CO	7.6	5.5	3.7	0.0002	---	---
CO_2	8.1	9.2	10.0	5.8	3.5	0.4
HCl	7.4	6.5	5.8	2.7	1.7	0.005
$Cl + Cl_2$	2.5	2.3	2.0	0.11	0.07	0.20
$H + H_2$	0.3	0.2	0.1	---	---	---
H_2O	10.6	10.1	9.6	4.0	2.4	0.34
N_2	44.3	47.8	50.8	67.3	71.0	75.9
NO	0.9	1.0	1.1	0.2	0.01	6.6×10^{-10}
$O + O_2$	2.3	3.0	4.0	15.7	18.9	22.9
Al_2O_3 (Solid)	14.0	12.5	11.1	4.1	2.4	0.3

* Stoichiometric equivalent is 1.41 lb of air for 1 lb of propellant

Water Pollution Assessment

NASA's Space Shuttle Programs may contribute potential pollutants to bodies of water in the following ways:

- Incinerator scrubbing effluent which may result in run-off of contaminated water to local drainage systems.
- In-flight failures which may result in vehicle hardware and possibly propellants falling into the ocean.
- Normal flight, which results in the impact of spent, sub-orbital stages (containing some residual propellants) and jettisoned hardware into the ocean.
- Eventual reentry of spent stages which have achieved orbit.

The problem of reentry debris is treated separately in this statement. On-pad vehicle failures would normally be expected to result in a fire that consumes most or all of the propellants, and, thus, be handled as an air pollution problem.

Spent vehicle stages which do not achieve orbital velocity are placed on ocean impact trajectories. In addition to stage hardware, small quantities of propellants (residuals and reserves) impact with the stage. These propellants are released and dispersed into the environment. Their probable effect on the environment has been estimated.

Vehicle hardware will normally sink in the ocean and slowly corrode. Isolated occurrences of floating hardware have been reported and provisions for tracking and demolition searches must be planned. In major part, such hardware consists of aluminum, steel, and fiber-reinforced plastics. There is a large number of compounds and elements which are used in launch vehicles in small amounts, for example, lead in soldered eleectrical connections and cadmium from cadmium-plated steel fittings. Neither the stage hardware or its corrosion products are belived to represent a significant water pollution problem.

The problem of water pollution relates primarily to the toxicity of materials which may be released to, and are soluble in, the water environment. A secondary consideration relates to oils and other hydrocarbon materials which may be essentially immiscible with water but, if released, may float on the surface of the water, inhibiting oxygen transfer, coating feathers of sea fowl, and fouling gills of fish. Solid rocket propellants are not sources of such materials.

Incineration disposal of by-products. - On-site incineration of processing waste materials will result in combustion product scrubbing to eliminate Al_2O_3 particulate and HCl gas in the disposal emissions. Propellant processing sites in Southern California and in Florida may be a part of the SRM booster program plan. In California, water quality in the Santa Ana River basin is judged by salinity and tested on the basis of Cl^- content. Evaporation and settling ponds of impermeable construction are acceptable for contaminated water handling. Quality of water used in the SRM booster program for Space Shuttle will be assured by recycling scrubbing water until settling and evaporation is advisable. The final step in the process will be to recover settled solids and containerize the by-product for transportation to approved solid waste disposal sites. The basic approach will be used at all processing sites pending coordination with appropriate local officials.

Normal launch considerations. - Potential sources of pollutants to the major pollutants are shown below:

Hardware	-	Heavy metal ions and miscellaneous compounds
Solid Propellants	-	Ammonium perchlorate

Jettisoned or re-entered hardware will corrode and thereby contribute various metal ions to the environment. The rate of corrosion is slow in comparison with the mixing and dilution rate expected in a marine environment, and hence toxic concentrations of metal ions (including heavy metal ions) will not be produced. The miscellaneous materials (e. g. , battery electrolyte, hydraulic fluid) are present in such small quantities that, at worst, only extremely localized and temporary effects would be expected.

The ammonium perchlorate in solid propellants is mixed in a rubbery binder and will thus dissolve slowly. Toxic concentrations would be expected only in the immediate vicinity of the propellant (within a few feet), if they occur at all. Consequently, a normal launch and flight will result in the downrange impact of spent stages and small quantities of residual propellants. The potential problem of a floating stage should be considered by analysis of tracking data and a surveillance flight to the impact area. Demolition follow-up on floaters appears to be adequate positive action.

Aborted flights and in-flight failures. - In the event of an in-flight failure during the early stages of flight, the vehicle might impact in the ocean intact, thereby exposing the large quantity of remaining propellant. Early in-flight aborts with SRM boosters have not occurred. With the exception of propellant quantity involved, the potential water pollution of an aborted flight is rated equivalent to normal launch and flight. Marine toxicity is not expected to increase by virtue of the SRM booster propellant ingredients selected. Demolition sequences, however, will require additional safety precautions.

Noise Assessment

Significant noise levels are generated in the operation of rocket engines and launch vehicles. This noise arises from the following sources:

- Combustion noise resulting from combustion in the rocket chamber.
- Jet noise generated by the interaction of the exhaust jet with the ground and subsequently the atmosphere.

- Combustion noise resulting from the afterburning of the fuel-rich combustion products in the atmosphere.

- Sonic booms.

Sonic booms have not been a problem in vehicle launches due to the location of the launch sites, the character of the vehicle trajectories, and probably due to the absence of aerodynamic lift surfaces. The impact of noise levels anticipated will be confined to the launch area.

Anticipated noise levels. - The major source of the noise appears to be the jet noise and its interaction with the ground and atmosphere. Both the acoustic power emitted and the frequency spectrum of the noise is affected by the size (thrust level) and the specific impulse of the rocket engine, as well as by design details. The nature of the noise may be described as intense, relatively short, composed predominantly of low frequencies, and infrequent. A range of 6 to 60 launches per year or 4 to 6 static tests per year are projected for SRM boosters on the Space Shuttle program.

The noise level for Space Shuttle launches using SRM boosters is expected to be equivalent to Saturn V launches. A summary of overall sound pressure level (OASPL) projections is shown in Figure 8. Both OASPL and maximum octave band sound pressure level data (OBSPL) are shown for Saturn V as a function of distance from the plume centerline. The expected OASPL for a static test of one 156-inch SRM booster is also shown in Figure 8 and is estimated from previous static test data of motors this size (ref. 10). Frequency spectra distributions of sound pressure level for Saturn V launches are shown in Figure 9. Note that the lower frequencies predominate and that the higher frequencies are attenuated more rapidly with distance. This indicates that the lower frequencies travel farther and "pollute" a greater area. These lower frequencies are less harmful to human hearing, and are less annoying (ref. 11), but are the prime cause of structural damage (ref. 12).

Environmental impact of noise. - Noise can affect man physiologically and psychologically. Physiologically, high intensity noise can cause permanent hearing damage and temporary threshold shift, i. e., the sensitivity of hearing is temporarily lowered. Psychologically, noise can create feelings of annoyance and discomfort in some people, while for other people the same noise can crease excitement and pleasure.

Research of the effect of noise on man has yielded criteria for noise levels and durations which man can generally tolerate. Table VII consensus values of the tolerance limits. The Damage Risk Values are thresholds beyond which hearing damage might occur. Comparing these values with the intensity levels of Figure 8 it is clear that within a 1-mile radius, intensity levels may be reached for a sufficient duration to cause permanent damage or temporary hearing loss of ear protection or shelter is not provided. Between radii of 1 and 2 miles, intensity levels may also be sufficient to cause temporary hearing loss and severe discomfort if ear protection is not provided. Beyond 2 miles, intensity levels will generally be found annoying and may cause momentary discomfort.

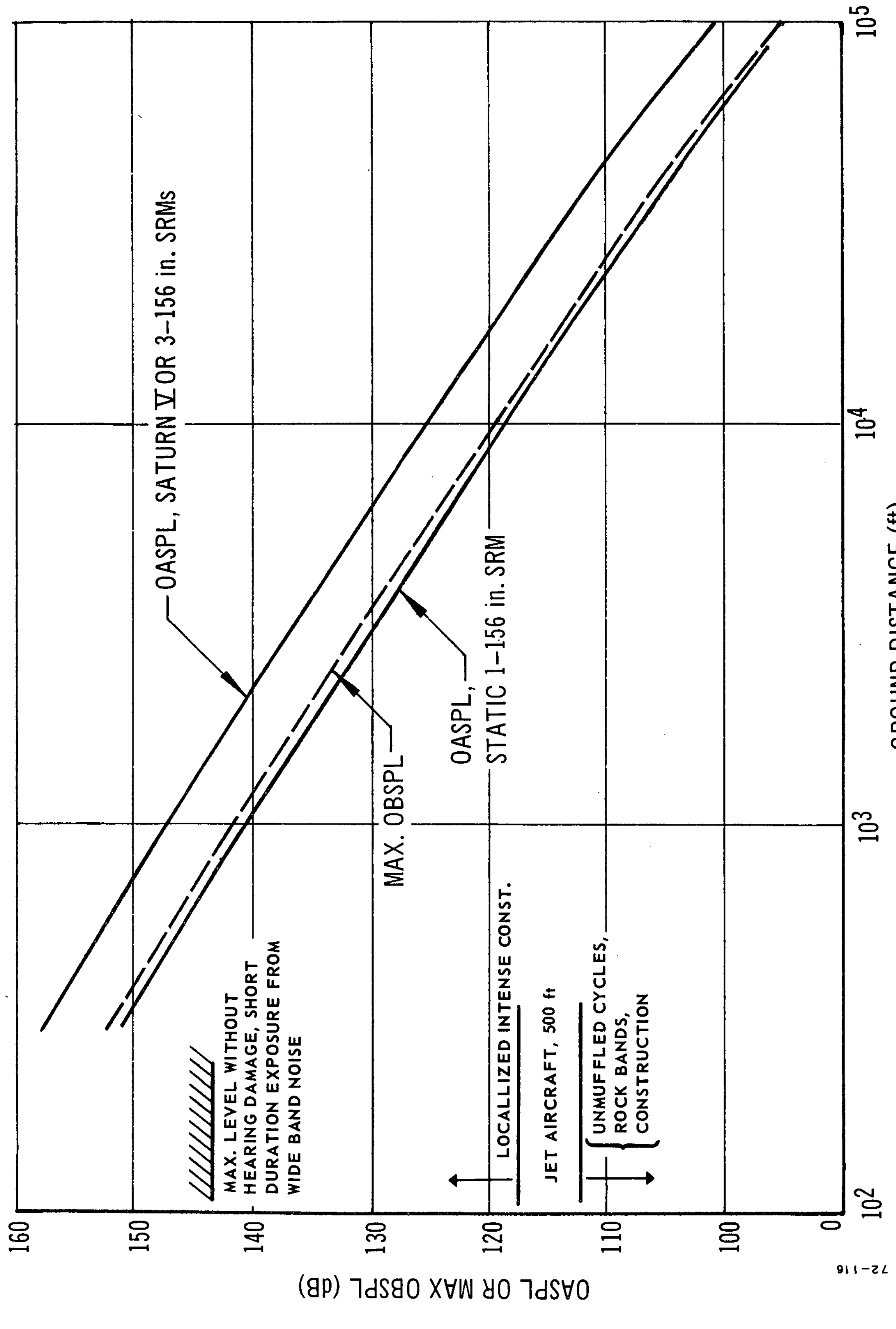

Figure 8. - Far Field Sound Pressure Levels for Saturn V

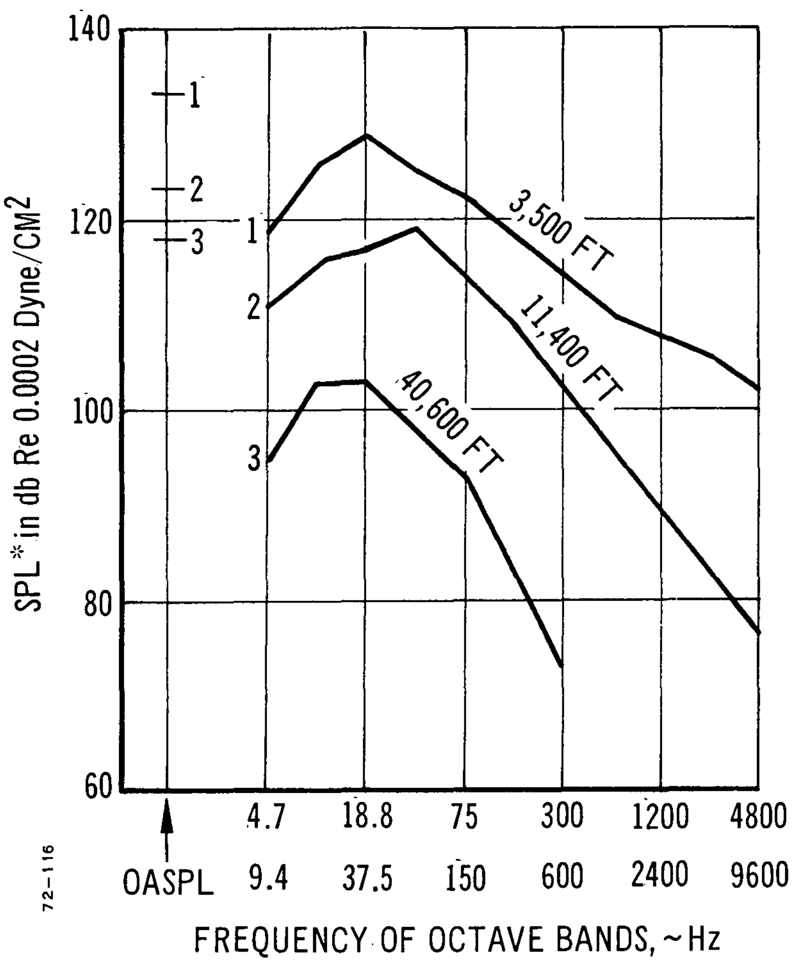

Figure 9. - Maximum Free-Flight, SPL Spectra for Saturn Test (ref. 13)

Figure 8 also contains indications of noise levels frequently experienced in urban communities.

TABLE VII

NOISE LEVELS FOR DAMAGE RISK AND ANNOYANCE
(refs. 11, 12)

Damage Risk Values (in db)	Annoyance Threshold	Damage to Ground Structures Threshold
142 absolute maximum value		
130 (10 seconds tolerance)	90 db (A)	130 db (frequencies lower than 37 Hz)
125 (30 seconds tolerance)		
120 (60 seconds tolerance)		

Structural damage is possible with high-intensity noise composed predominantly of low frequencies. Comparing the damage criterion shown in Table VII with the intensity levels listed in Figure 8, structural damage would not be expected outside of a 0.5-mile radius from launch. With appropriate structural materials and techniques, damage within short distances of the launch pads, all within controlled areas, can be avoided.

It is clear from these data that for any single rocket booster test or launch, "noise pollution" occurs over a relatively wide, but controlled, area. However, with its short total duration of 3 to 4 minutes, its frequency occurrence, and the imposed safety precautions, the boost noise is not considered to have a significant impact on the environment. No uncontrolled areas are close enough to the launch pads or static test facilities for any significant effects to result from exposure of the public or uncontrolled-area structures to these noise levels.

ALTERNATIVES

The activities which contribute to potential environmental impact are the development and static test firing of the SRM boosters and the launch of the Space Shuttle vehicles. The matrix in Table VIII displays some of the alternative actions which might be taken in these areas. At present the analysis of environmental impact does not disclose significant disadvantages to the SRM booster approach. The most objectionable emission produced is HCl which causes a localized and controlled impact. Consequently, development of "clean" solid propellants may be an attractive alternative. Tailoring of the solid propellant formulation with the specific objective of reducing HCl may be possible, but will require nonrecurring development cost in the range of $50 to $100M. Successful tailoring will result in decreased performance and consequently, larger SRM booster stages. The potential tradeoffs available are illustrated in Figures 10 and 11.

The performance reduction of SRM boosters as a result of potential HCl reduction approach is shown in Figure 10. Substitution of nitrate containing solid oxidizer, for ammonium perchlorate can achieve 40 percent reduction of HCl on a theoretical basis with a 1 to 3 percent performance loss. In the case of $NaNO_3$, the substitution results in appearance of NaCl in exhaust products. Development is required in the use of mixed oxidizers. It also is possible to reduce particulate Al_2O_3 at the cost of performance as is shown in Figure 11. The drawback to each approach discussed is that complete reduction of the objectionable specie cannot be obtained within any realistic performance envelope. Further, the presence of these species causes little environmental impact under the projected use of SRM boosters. There is little impetus, therefore, to pursue these alternatives. Finally, it should be noted that the alternative of LOX/LH_2 stage, which does have capability to eliminate the objectionable emissions, cannot provide thrust-to-weight ratios suitable for booster service.

TABLE VIII

MATRIX OF ALTERNATIVES

Activity	Alternatives: Use of Scrubbers, Collectors, etc.	Use of Remote Sites	Development of "Clean" Solid Propellants
Research, development, and ground test	Potentially complete elimination of objectionable emissions. Practical only for small scale evaluations.	Already reasonably remote. No effect on global basis.	Reduction or elimination of HCl emissions.
Launch	Practical only for first few feet of flight.	Already reasonably remote. Few alternative sites available. No effect on global basis.	Reduction or elimination of HCl emissions.
Comment	Increased development and operational expense. Modest overall reduction in emissions. Present analysis discloses no significant impact of present emissions.	Extremely expensive.	No formulations known with performance comparable to current solid propellants. Tailoring possible for HCl reduction (but not elimination) at nonrecurring development cost of 50 to $100M. Impact of HCl localized and controlled.

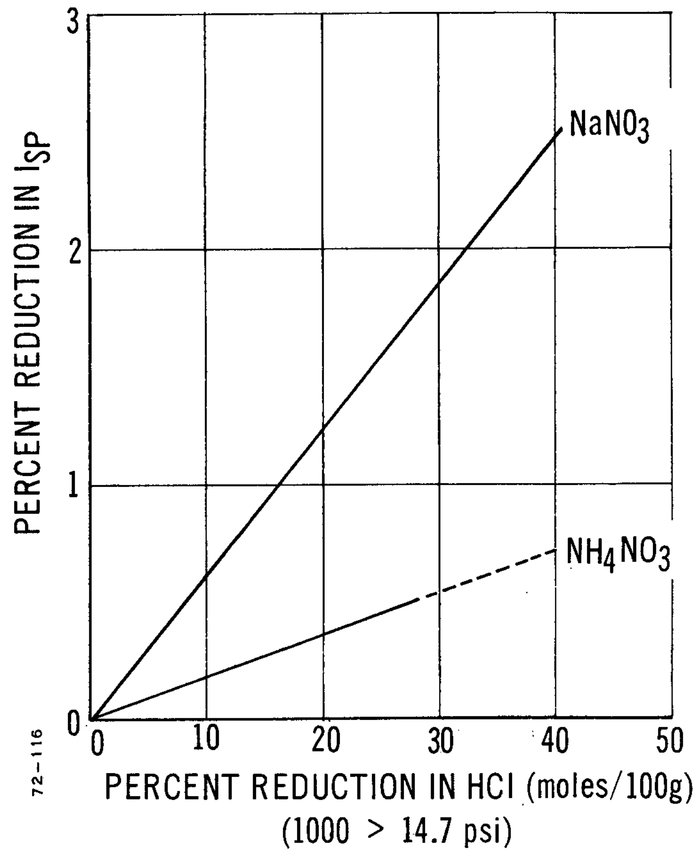

Figure 10. - Performance Reduction as a Function of HCl Reduction in the Volume

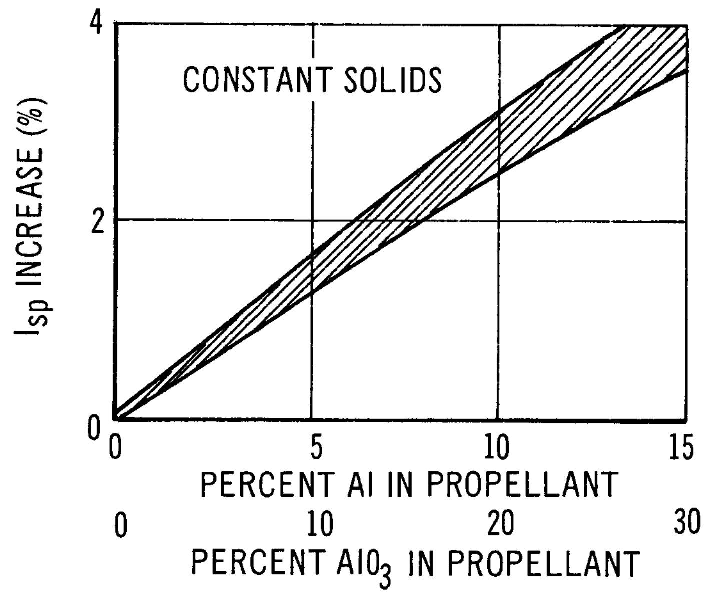

Figure 11. - Performance Tradeoff, Particulate versus Al Content

LOCAL SHORT-TERM USES OF THE ENVIRONMENT COMPARED WITH LONG-TERM BENEFITS

The comparative relevance of specific space program objectives to broad national goals has been investigated recently so that long-term space transportation needs can be determined realistically. Such planning endeavors have helped to identify and document the value of space programs in relation to man's historical need to better understand, protect, and control his total environment (refs. 14, 15).

The general value of past or planned space activities can be simply expressed as follows. Scientifically, more has been learned about our immediate environment and that of the solar system since the inauguration of the space age than in all previous ages combined. The knowledge obtained is fundamental to practical programs of environmental protection. Noticeable improvement is being made in using acquired space capability for such functions as communications, navigation, and meterology. The current NASA effort (OSSA) in the area of orbital earth resource surveys is particularly significant regarding long-term environmental productivity. This effort has a unique potential for providing man with an operational capability to measure, monitor, and manage environmental conditions and natural resources from a local to a global scale.

The Space Shuttle program concerns payloads which have no environmental impact aside from that associated with momentary impact of static test, processing operations and the launch process. These payloads are expected to provide long-term benefits to the Earth, its environment, and inhabitants.

IRREVERSIBLE AND IRRETRIEVABLE COMMITMENTS OF RESOURCES

Irreversible and irretrievable commitments of resources are difficult to assess on a program which involves only a few minutes of activity at infrequent intervals. The energy release from SRM boosters for a Space Shuttle launch represents an extremely rapid conversion of chemical energy into heat. However, both totals and rates of energy release are insignificant by comparison with electric power generation from fossil fuels. The average consumption of fossil fuels for U. S. electric power generation approximates 4×10^8 Btu/sec compared to an SRM booster energy release rate of 7×10^7 Btu/sec. More importantly, the energy release for the SRM boosters occurs only for a period of about 2 minutes. The consumption of fossil fuel for power generation continues on an uninterrupted basis.

The materials that make up the Space Shuttle launch vehicle ready for launching are largely irretrievable once the launch process is initiated. However, the resources that are used are replaceable from domestic resources with relatively insignificant expenditure of manpower and energy.

The largest weight of materials in SRM boosters are the propellants. These common chemicals have previously been enumerated and defined. Resources and energy required for their production are not significant in comparison with, for example, the resources and energy required to produce the current production rate of 1 million barrels of jet fuel per week for private, commercial, and military jet aircraft.

The next largest amounts of materials are iron and aluminum. Other materials include plastics and glass, as well as other metals such as nickel, chromium, titanium, lead, zinc, and copper. The quantities of the inert materials described are insignificant in comparison with those used in one year of production (10,000,000) of automobiles. Further, much of the material used for automobile manufacture is not returned for recycle representing an irreversible and irretrievable commitment of resources in another sense.

The largest fraction of SRM booster expenditures are for wages and salaries. These expenditures represent a very small fraction of the national economy. Consequently, commitment of resources to this program is expected to have a small but positive effect on the national economy.

DISCUSSION OF PROBLEMS AND OBJECTIVES

Problems

Certain problem areas of a selective nature have been identified in the course of preparing this environmental statement. These problems are associated with a lack of precise knowledge of launch vehicle emissions and difficulties in identifying precise consequences of certain events. However, in no case is it anticipated that more complete knowledge would alter the conclusions expressed in this statement.

Objections

No objections have been raised to the statement at the time of this draft (February 4, 1972).

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